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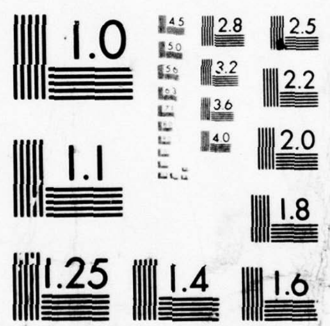
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PROCEEDINGS OF AFFDL FLYING QUALITIES SYMPOSIUM HELD AT WRIGHT STATE UNIVERSITY 12-15 September 1978

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Interim Report – August 1973 - September 1978

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Proceedings are reported of a symposium/workshop held at Wright State University, Dayton, Ohio during 12-15 September 1978. The symposium was sponsored by the Air Force Flight Dynamics Laboratory as part of an effort to revise MIL-F-8785B, "Military Specification, Flying Qualities of Piloted Airplanes." Flying Qualities specialists were brought together from industry, government agencies, and university. Topics for discussion were 1) experience and problems with MIL-F-8785B, 2) revisions to MIL-F-8785B currently proposed by AFFDL/FGC, over

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3) a new format for the flying qualities requirements, and 4) future requirements in general. Full-length and informal papers were solicited to address any of the preceding topics. In addition, working sessions were organized to discuss the current revisions and future requirements from the viewpoints of different technical disciplines. This report contains the papers and summaries of the working sessions, as submitted by the authors.

FOREWORD

This report contains the proceedings of the AFFDL-sponsored Symposium and Workshop on Flying Qualities and MIL-F-8785B, held at Wright State University on 12-15 September, 1978. The papers contained herein were prepared by various authors. The report editors were Messrs. G. Thomas Black, David J. Moorhouse, and Robert J. Woodcock, of the Control Dynamics Branch of the Air Force Flight Dynamics Laboratory (AFFDL/FGC). The symposium manager was Mr Black.

This work is the interim report for the time period of August 1973 through September 1978. It was performed under Program Element 62201F, Project 2403, task 05, work unit 19.

Special thanks go to the Media Center, the Engineering Department, and to Lorna Dawes and the Office of University and Community Events at Wright State University. Their assistance and support of the symposium were greatly appreciated.

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SECTION I

INTRODUCTION

INTRODUCTION

MIL-F-8785B,"Military Specification, Flying Qualities of Piloted Airplanes" was issued in August 1969. Together with amendments dated March 1971 (interim) and September 1974, this document presents the current generally-applicable flying qualities requirements for US military airplanes. Many requirements soon became less applicable because of advancing technology, particularly the use of the flight control system to modify classical airplane responses. A revision effort was conducted within the Flight Dynamics Laboratory from 1973 through 1975. After an internal and a government-agency review, the proposals* were issued for industry review in early 1978.

An AFFDL-sponsored symposium/workshop was held to bring together government and industry representatives directly connected with development, use or application of MIL-F-8785B. As an integral part of the review cycle, a primary objective of the symposium was to solicit comments and objections/endorsements to the proposed revisions. Formal and informal presentations were solicited on both MIL-F-8785B and the revisions. Working sessions were also organized to discuss the revisions from three technical viewpoints: design, flight test and analytical methods. The comments and discussion will be considered in drafting an Amendment 3 to MIL-F-8785B. When Amendment 3 has been coordinated and published, it will be the last version in the current specification series. After that a change in format is planned to a Prime-Standard with a back-up Handbook. Such revisions are in prospect for all military specifications, with the object of reducing confusion and eliminating unnecessary requirements. The Standard will typically have blanks in place of numerical requirements, while the Handbook will contain the information to fill in the blanks for a particular airplane. It will have justification for each requirement, recommended criteria (maybe with alternatives) and

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*"Proposals for Revising MIL-F-8785B, 'Flying Qualities of Piloted Airplanes,' " AFFDL-FGC Working Paper, February 1978.

substantiation. A second objective of the symposium was, therefore, to provide initial inputs to the longer-term revision process. Since the current revision items only form a partial list, part of the task was to identify remaining deficiencies in the present specification. Comments were also solicited on future requirements. Finally working sessions were organized specifically to consider the future.

This report presents the papers as submitted by the various authors from both industry and government. In addition, government moderators were appointed for each working session to record the comments. Summaries of the significant discussions and opinions in the working groups are presented as submitted by the moderators. Editorial comment is presented in Section X, Summary and Concluding Remarks. Finally Appendix A is a list of attendees with affiliations, and Appendix B is a paper not presented at the symposium but judged to be appropriate, relevant to some of the discussion concerning flight testing to the requirements.

SECTION II

REMARKS

Welcome and Introduction
Mr. W. E. Lamar
AF Wright Aeronautical Laboratories

It is a pleasure for me to welcome you to this seminar and workshop on behalf of AFWAL, the Air Force Wright Aeronautical Laboratories, and also the Air Force Flight Dynamics Laboratory.

It is my pleasure this morning to introduce a man who is eminently qualified to open up this meeting. He's been involved with flying qualities from many different viewpoints over his entire career. He is a military pilot, and one of the first astronauts. He flew cargo aircraft over the hump during World War II, F-80's during the Korean War, and F-4's with 189 combat missions during the Vietnam War, so obviously he understands the flying qualities of combat aircraft.

Some twenty-five years ago he was much involved in developing and flight-testing automatic flight-control systems in the Directorate of Flight and All Weather Testing of the Wright Air Development Center. As a test pilot, he has flown both advanced military aircraft and advanced research aircraft. He is probably the only pilot in the world who can tell you what the trim and handling quality changes are when a nose landing gear inadvertently extends at Mach 5. That happened to him when he was flight testing the X-15 Rocket Aircraft.

He has been the Commander of both the AF Flight Test Center at Edwards and 4950th Flight Test Wing at Wright-Patterson. He also has been the Commander of the AF Test and Evaluation Center which evaluates the total capabilities of weapon systems. So, we have here a man who has had a wide range of personal experiences in the development, flight test and research test of flight control systems plus extensive flight experience, both in peace and war, with the flying qualities and related aspects of military aircraft. And, as the Vice Commander of the Aeronautical Systems Division he is still very much involved. It is a great pleasure to have here this morning Major General Robert Rushworth.

Opening Remarks

Major General R. A. Rushworth

Vice Commander, Aeronautical Systems Division

Thank you, Bill. Good morning ladies and gentlemen. It is a pleasure for me to welcome this group and open this symposium. As Bill says, I've been involved with this a long time, as I guess I've seen it from all directions, having worked on papers to go into the handbook, having worked on airplanes to satisfy the specification, having rewritten things that go into airplanes, and a variety of different things. I certainly didn't think when I started out some twenty-five years ago that I'd be up here telling you people that maybe we didn't do it all right then but there has been a change in the overall environment.

Your workshops are tailored and you're zeroing in on one specific thing: how to revise a particular specification that is very important to every one of the people who get involved with flying within the military. It's equally important for the Air Force people, as it is for the Navy, the Army, the contractors, NASA, civil service, wherever it is, and also translates into the total civilian community regardless of whether it's light airplane flying or large airplane flying. I think what we've done in the past, things that we've dictated have translated very well to help solve some of the problems, but I'm not sure that in the past even though we've had the flying qualities, 8785B, and they've been revised and updated and whatnot, that the specification has been totally adequate. I say that from both sides of the issue. It hasn't been totally adequate to satisfy the designers of the system. I think in that sense it's important that you do work here in coordination with all the people in the community to try to satisfy both of those conditions now.

Perhaps one of the questions that I could raise that I think you might be working on is (and it's a very simple question), "Have we had

recent change in the environment?" I personally think that we have. I've seen a trend over the last few years wherein people are trying to do more with airplanes than we've thought possible; not that we haven't wanted to do it before, but we knew that it was just impossible. But now we find people doing things with airplanes that we don't think they should, and they want to do it. The operators want to do it, because it's important to them in the roles and missions that they have to carry out. I think that should be paramount in your thinking as you go through these particular workshops and through all of the subjects in that, again, I think that we're asking more of older airplanes now than was ever conceived before. I'll give two specific examples, that show what the criteria are. The F-4 is an extremely good airplane brought out by the Navy some 15 or more years ago to do a specific job. When we got that airplane into Vietnam both the Navy and the Air Force were using it in completely different roles than for which the airplane was originally designed. We lost a goodly number of airplanes because we had not adhered to the total handling qualities/flying qualities within the specification of the specific airplane, and the airplane just was being directed, asked to do much more than we ever envisioned, originally. I can bring up another fighter-type example. Although it's a trainer, the T-38 is being asked to do much more today than was ever conceived of it when it first came out back in the early 60's. TAC is using it as a trainer. It's a very good trainer in that we can train young pilots in their initial flying careers, but it's being used now as a trainer in gunnery. It was not conceived for that, and it is not a particularly good airplane, as far as handling qualities, for that, but it is the best that we have and it's one that hopefully we can keep around for some time. But you'll see as pilots transition they do have to have better equipment in order to do their particular job. The other big change is the B-52. We're going to be asking more of that airplane, and it's probably the oldest thing that we've got in the inventory that we're working on right now. We're going to be asking more of that airplane in the next few years than was ever conceived

before. So a problem that we see is adapting old systems to the new environment and adapting new systems to the changing environment; we think it's a great problem. We hope that you can address that throughout your meetings here.

I might raise another consideration: I think in specific instances we have systems on board today that can easily be better in the handling qualities, the flying qualities, than the specification requires, and I think the F-15 could be an example. The F-16 certainly is an example. We're well into the testing of the F-16; we've found some different things and we've certainly been pleased with what we've seen, although with some pleasure there's also some concern. That airplane is going to have some limitations and restrictions that are within the flying boundaries that the pilot wants to operate in. That's to us a very serious consideration, but not to everyone. Everyone thinks that airplane is going to have all of the qualities, and the best qualities, that could ever be put into a system, and that's true. But it is going to have limitations because we can't insure that those pilots don't cross-control, don't get their left foot on the right rudder pedal, that sort of thing; sooner or later one of them is going to tip it off and it just won't return to normal flying. So we have concerns and as I say, there are systems right now that can do as well or better than the specification directs.

I'd like to transfer over to what I was looking at in one of your sessions here: session number two, "Special Problems 1." If I go down and look through those nine items, I think I can sum up in one word what probably is the most significant problem in our business today, and that is communications, or the lack of communications. I don't think it's difficult for all of the people to talk to each other in this room and be able to adequately address what you're saying and make the other individual understand it. That's not where the problem lies, and in prior years, perhaps, that's where we've stopped. We've always known that we could design an airplane and design a system to go into that airplane, put it out in the field, and expect it to work 90% of the

requirements using command. What we've got to insure now is that the operator, since he doesn't have quite the effective communications capability as all of you people in this room, gets his opinion across, his requirement, his desire. Then it's going to be up to us and to you people in particular to make sure that what goes into that airplane, the initial design, the qualities, and what adds on as a system within a system, is going to be effective for the new generation of vehicle that comes out.

I'll talk on just a brief bit, about the future and something that's coming along: MIL-PRIME. I don't think in reality that MIL-PRIME is going to have a great effect or great change on any of the systems that we're conceiving right now. It perhaps, in a certain sense, may make it easier for the contracting community, the design community, in order to do their job, but the overall result is still going to be that we must satisfy the operator. We must satisfy his requirement and in certain cases we won't know that until the airplane is pretty well designed.

I'll leave you with one simple message: Hopefully we can get vehicles and systems that have good responsiveness, good repeatability and good reliability throughout the total flight regime. If you can do that, you'll have solved the problem; we won't need a symposium next year, and one handbook specification will take care of the whole issue. Thank you very much. Have a good day.

Introduction

Mr. W. E. Lamar

AF Wright Aeronautical Laboratories

I wish General Rushworth could spend some time with this meeting, because I'm sure he would contribute in many areas.

Next we're priviledged to have with us, and he's going to be here most of the day I understand, a man who brings a different perspective to the meeting. He's from Washington, he is in the Pentagon, and he is essentially a deputy assistant secretary although his title is a little different. He is a very strong man in electronics, and this meeting is blending electronics and flight controls. He's been a professor in that area at a number of different universities, and he's also been involved in the electronic industry. He came to the Air Force from IBM, so he's very well up-to-speed in all these areas, but he brings some distinct qualities. Since I've known Walt he has a very strong flavor for precision. He has a tremendous capacity to delve into the fundamentals, into the basics, and I'm sure while he is here he will task you wherever he sees it necessary to get down to the basics and make sure your thinking is clear from the fundamentals on up to the problem that you're considering. So it's a distinct pleasure this morning to also be able to introduce, to give a few words, Dr. Walt Beam, the deputy for advanced technology, working for Dr. Martin who is Assistant Secretary of the Air Force for research, development, and now logistics.

Dr. W. R. Beam

Office of the Assistant Secretary

Department of the Air Force

Thanks, Bill. It's certainly a pleasure to be here. I recall with pleasure my attendance at the stall-spin workshop several years back, which I believe was mostly the same group of people. As Bill Lamar implies, I have different credentials. I maybe have one tenth of a flying hour for every flying hour that General Rushworth has. His comment about the right foot and the left foot with respect to fighter pilots made me realize all of a sudden what the control stick was for: it was so they couldn't get their feet crossed.

I would like to welcome you here on behalf of Secretary Martin, and I speak to you mostly I guess as an amateur pilot. I think I have now about a hundred hours at all types of controls, power and sailplane. I'm building a sailplane at home in my spare time, and there's the lacquer-primer-surfacer under my fingernails to prove that I'm really getting along with it. I expect roll-out any day now.

Flying is, as I think most of you who have been involved one way or another are aware, a personal experience. It is highly subjective. Some people make a lifetime career of it and are really thrown into the depths of despair when by dint of being promoted to Colonel they are taken away from it. Others, like Harrison Schmitt, the former astronaut and now senator, learned to fly and learned to fly very well for getting one thing done: to get on to the Apollo shot, and Harrison has not flown since. So people are different.

Fortunately, for pilots there is more similar than there is different about flying different aircraft. Every time I get into a new sailplane I always worry, "Is this thing going to behave strangely different from what I'm accustomed to?" Fortunately, it isn't all that different until you get going very slowly, until you start approaching the ground and you pull out the spoilers and the tail hits the ground first and you do that sort of thing. It's the secondary characteristics in which aircraft differ. Unfortunately, more people are killed due to secondary characteristics than

due to primary characteristics: you don't stall unless you're going slowly, things like that - or unless you're in a wind shear. I think one of the things that is characterizing this flying qualities business is that you are paying more attention to secondary qualities, having gotten your arms around the primary ones pretty much with the previous specification that you put under your belt, which has been very successful. The F-16 I know benefitted substantially from having the flying qualities criteria around, and other aircraft will also.

We're now entering an era, as Bill Lamar implied, in which digital fly-by-wire (though of course the F-16 has analog fly-by-wire) is the coming thing, not for the reason that everybody wants to have a computer, but because digital systems, once you get the interfaces hooked up, are much more precise than analog systems. They do not get out of adjustment. The software, if it's wrong, stays wrong until it's fixed, but if it's right it always works (which is no great satisfaction to a lot of people). We're also, with this digital fly-by-wire era, entering a situation in which no particular relationship need hold between the controls and the response of the aircraft to those controls, which says that you've got a clear slate as far as the things that you can do with the airplane. On the other hand, you must have certain constraints, or else the guy who flies the thing isn't going to know what is going to happen when he moves the controls. So there's a very delicate nuance between retaining the control response that one expects out of an aircraft and providing some of the special modes in which the aircraft may fly up, down, or sideways.

We're also entering, of course, an era in which many aircraft (particularly fighter aircraft) will have more useable dynamic degrees of freedom than in the past. Of course, we've had flaps and slats and various things for a long time, but we have not had anything other than primary controls for the three axes of attitude control which could be operated in a dynamic way, that is, swished back and forth. Let me tell you a funny one. When I first rode on a T-39, which is approaching four years ago, I wasn't aware that the slats were automatic. When we were taking off, just about the time of liftoff this thing started going like that (shaking) and I

said, "My God, what kind of a pilot do I have up there that his hand is moving that fast? I need someone with more experience than that." It turns out that the wind was doing that, not the pilot.

By the mere process of putting control surfaces on both ends of the aircraft we create a CCV or an AFTI, and enable ourselves to get both direct lift and direct side thrust, and by putting enough of them, or by putting big brakes out there, we can get direct stopping or nearly so. There's a considerable amount of debate as to how much of that, putting on the brakes, one wants because there's a school that believes in the energy maneuverability theory where energy is money in the bank, and the more drag the less energy, and so on. We have not sorted out, I think it's fair to say, the relative importance of being able to slow down in a hurry versus maintaining your speed. There has been some work with Harriers in which the Harriers have escaped from faster, more maneuverable aircraft by the process of cranking the nozzle down, and I think you can crank it slightly forward if I'm not wrong, and essentially slowing themselves down or lifting not straight up, but having an unexpected rise in the position of the aircraft as compared to a guy who is flying along behind them. Some of this, putting control surfaces on both ends is bound to remind someone of what they did when they first built aerial ladder trucks for the fire departments. What they did is they put a tillerman on the back end, and he steered his end and the other guy steered his. I think the stage of the game we're at with CCV's and AFTI's and such is probably pretty much at that primary point. We don't have a two-man aircraft with one guy steering the empennage and one guy steering the canards, but it is not developed terribly much farther from that in terms of where it will get to be.

Despite all of our technological advance, some things remain inadequately understood, particularily high-angle-of-attack behavior in which one always, when one buys an airplane or picks up an airplane, looks at the book and looks at the little section entitled "Spin Recovery Techniques." It will often say "spin recovery is normal" which does not give much confidence to someone who is flying an airplane that is said to have viscious spin characteristics for the first time. There are people who argue that

the best technique is to push the opposite rudder hard over until all rotation stops and slowly but purposefully push the stick forward until flying speed is obtained, being careful that one does not exceed redline speed in the pullout. If one is having an incipient spin at 500 feet I don't think one has very much time to go through that "slowly but purposefully" type of deal, and one doesn't have time to bail out, either, at least in the airplanes I fly.

So there are things that are by no means well understood. Whether indeed we will get our secondary characteristics, high-angle-of-attack characteristics, really worked out to the point at which the pilot will have a sense that he is in control of the aircraft through a much wider range of maneuvering than he presently does remains to be seen, and I mean the average pilot, not the test pilot: test pilots for the F-16's and such can put them on their tail and skid along with the thing at 55 degrees, and I think have a pretty good idea as to whether it's going to fall off to the side; but your average pilot is going to be told to stay out of such regimes because he won't really know what to do when he gets into them, and he won't have a spin chute in the back in case he runs into trouble. I think we can probably improve in that situation because if we have more control surfaces we will have more control area, and control surface area has a lot to do with getting out of trouble, so there will be new things to learn with the six-degree-of-freedom aircraft about their departure characteristics.

The digital flight control system business is really part of a larger movement, a movement that spans the whole area of the aircraft business. In the avionics area, which I'm sad to say is distinguished from the flight control area and from the power area (mostly by barriers of Laboratory or divisional or branch separation) digitalization is taking place at a rapid rate for several reasons: one, the natural precision of digital systems; and secondly, the great flexibility one has in changing the systems if one finds that they are not right. However, with change comes the opportunity to make change that is not needed, and that is one of the things that happens when you set up a hundred man programming group to support

an airplane. We have to learn how to control ourselves there. The digital flight-control system business offers opportunity. One of the reasons I'm telling you about these things, which are sort of ahead in the game, is that as you develop a new flying qualities specification it is very worthwhile to keep in mind what's going to happen next so that as you make definitions, as you establish criteria, you can say, "How will that apply if there is one more degree of freedom?" In other words, you very often find yourself in the position of (if you don't think into the future) establishing something that you'll have to change completely because it will no longer be valid, whereas if you said it a different way or established a criterion in a different way it would apply despite the addition of new technology. For example, digital flight controls afford the opportunity to make the controls non-linear as far as the response of the aircraft to pressure or motion of the controls is concerned, and this has been used. I believe the F-16 has sort of a quadratic or a segmented quadratic roll rate in response to sidestick roll pressure. People who talk about non-linearities in the flight-control system, however, I think are careful to leave them at the pilot-control interface, and not crank them down into the guts of the servomechanism, digitalized or analog, which is their interest in making sure that they haven't put any instabilities into that thing as far as they're concerned. If the pilot wants to have pilot-induced oscillations because of the way the controls work, that's his problem. So it figures that as you introduce non-linearities on purpose into systems you are going to have to be responsible for characterizing their effect on the linear or non-linear pilot - and not just the experienced pilot who can quickly size up the situation, but the low-time pilot who's transitioning to the aircraft. I noticed that although we apply this quadratic type of thing to the roll, we don't do it to the pitch, because the last thing anybody wants is to have a pitch that goes crazy if you push a little too hard forward or push a little too hard back; you don't want your g's to come on in a big hurry. People like to have linear g versus stick force; they've been happy with that. I think they know what it feels like. We also don't do it in yaw; I think the main reason is that people in jet airplanes don't use their feet very much, except in landing events

occasionally, and some squirrelly maneuvers, I suppose, trying to keep the thing from sliding off in the wrong direction. But I'm not sure that most jet pilots really do practice using their feet very much, so maybe it's a matter of "don't care." The assumption always is that if there is a nonlinearity of any sort the pilot is able to compensate for that. You are giving him a total credit, which is right, but you're also giving him perhaps an additional job which you should assess as to whether it's good or bad. I think the business of nonlinearity in roll is very similar to the nonlinearity in steering a car where you don't have to turn as far to get the last bit of lock as you do at the middle, so it makes a lot of sense.

The pilot as a servomechanism has been modeled to some extent by the Aerospace Medical Research Laboratory. The thing that most impresses me about their results is "by golly, he's got a certain linear servo-type of thing with a very nice cutoff frequency characteristic and, by golly, you better not build a system which with that pilot in that loop is going to have an instability problem." I haven't seen anybody who has looked at these models of the pilot (it may be done because there are people doing everything down in the works), and added the pilot's phase characteristic and amplitude characteristic to that of the rest of the system and said, "What are the likelihoods of pilot-induced oscillations?" We don't really know what is best in the way of handling qualities. I don't know what you guys think of as flying qualities versus handling qualities; somehow handling qualities implies subjective character, the flying qualities as reflected to the pilot. The airplane may fly fine, but to the man who's handling it, it either handles nicely or it doesn't. We think that in calibrating an aircraft or in measuring an aircraft subjectively, the "fussier" the adaptation of the pilot to the aircraft, the less good the aircraft is deemed in terms of a "pilot's" aircraft. While I'm not one to cater to any minority group such as pilots, nonetheless, they've a lot of other things to do, and if the airplane is a dog to fly, they're going to spend more of their time flying the airplane than they really should. On the other hand, if you take away all the feel of the airplane, there are certain things that they're not going to be able to do in terms of

maneuvering that aircraft, and this is part of your job, in the next years: to try to understand better, through simulation, through varying the parameters in some of these fly-by-wire aircraft.

The digital flight control system lets us build in limits to control authority, and one of the things that we still have not answered and cannot really answer at the moment is, "Should we build in these limits so that the pilot can't hurt himself or the airplane if the sensors on the aircraft know that he could do it?" In other words, should the thing absolutely come to a stop before he wrenches the wings off or should the pilot have to limit himself in some way? I think probably the answer is that if we can build the digital flight control system so that the pilot can throw the thing around at will, that it's better. In other words, if it can be accurate enough that he can get as far out as it's safe to go, we ought to let him do that, but the trick is in determining how far out is safe. This is a question that we have to answer, obviously, in the fly-by-wire systems.

Another area which is principally of concern in the Navy is the vertical take-off and landing business, which established brand-new limitations on what are normally thought of as aircraft. Obviously, the stability criteria of the Harrier when it is in a situation in which the wing is not doing very much lifting are totally different from the stability criteria when the thing is flying forward, and you've got the stabilizing airfoil lift; and from the rash of accidents with Harriers which was explained as low-time pilots it is certainly evident that it is not a terribly simple airplane to fly. It's not at all clear that this handle that cranks the nozzle down is the most ideal way of changing the direction of your thrust, or is in any way related to how the pilot would think if he thought of cranking the thrust down. It is obviously a difficult problem. Each type of vertical-takeoff and landing aircraft has entirely different stability criteria depending on where its thrust is, and where the thrust-carrying things are, and God help us if the engine fails, but we mostly watch the Navy in that area.

Probably the most interesting area to me is the area of six-degree-of-freedom aircraft. This is a term which we blithely use to describe an aircraft which has got as a minimum some canards that can twiddle the front end of an airplane and a speed brake that can slow it down so that you get all three attitudes and all three displacements - not necessarily perfect displacements; you may not want perfect displacements, but nontheless perarately controllable actions. Clearly, having separate controls on the forward canards from the tail would provide a situation like the aerial ladder where the guy has two hands, one running the front and one running the back. We're smart enough not to do that. We have thus far, I think, with the CCV vehicle, put direct-lift function onto a thumb-switch on the good old control stick, and of course, the control stick by my count has at least five other functions on it, so it's not at all clear that that is the place to put additional functions. I keep wondering whether with the six-degree-of-freedom aircraft we might come up with a more subjectively realistic control thing in which the control stick has more degrees of freedom than does the one that we presently use, in which you could both push it to one side and twist it. I'm not sure that the muscles in the hand and the arm are up to doing what you would have to do in order to control four degrees of freedom with the one stick, but certainly there are some combinations that have to be looked at in terms of getting the most natural ability to fuselage aim or to slide nicely to the side, which is done with the feet - and I think that's probably pretty natural because people are accustomed to doing slips and yawing the airplane, and that comes naturally. The pilot workload is critical: if the six-degree-of-freedom aircraft requires more concentration on the part of the pilot to do his job it's obviously going to be less successful in the sense that in watching out for missiles, other airplanes, and so on, there won't be as much time and concentration for that.

The business of multiple modes is obviously something of much importance in a fly-by-wire aircraft, particularly six-degree-of-freedom aircraft. I'm not at all sure that we have optimized what these modes are: constant-attitude modes, fuselage-aiming mode, whatever. We seem

to be guided more by past history in autopilots, which I don't think is pertinent, because autopilots really are long-term things where you adjust the knob to make a turn, and there are no real fast dynamics, whereas the six-degree-of-freedom controls are only to be used in a high-dynamic situation: aiming to the ground, following another airplane, and so on. If you take, for example, the most extreme case of six-degree-of-freedom aircraft which has integrated fire and flight control where you are trying to do aerial gunnery against another aircraft which you are chasing, and there's a radar that can tell you where the other aircraft is (or a laser) and there's a radar that can tell you what range it is, what then does the pilot do in that aircraft if there is servo system that can follow the other aircraft around? One of the answers, of course, is that the only thing that those sensors can't do is to tell what the banking characteristics of the airplane you're chasing is, in other words, whether he's going into a bank, whether he's going to turn. Obviously, then the pilot is the good sensor of that, and you want to hook him up to the controls so he's able, possibly with some lead functions or whatever, to do what he can do best, in order to couple the automatic part of the system to the manual part.

So there are some very challenging jobs ahead. I think we must always apply a final criteria: does the pilot feel that the airplane is doing a capable job? Will it do what he needs done? If the answer is not "yes" on that (it's partly subjective, admittedly) then I don't think you've succeeded. I'm reminded of the rather silly thing which is nonetheless true: the public response to two automobiles with the same engine, one with a light spring under the accelerator and one with the heavy spring under the accelerator. Everybody says that the one with the light spring is the more powerful car. If you can figure that out, you can probably figure out the airplane business, so I hope you'll carry on successfully here today. Thank you.

SECTION III

SESSION 1: A REVIEW OF THE CURRENT SPECIFICATION, ITS USES, AND PROBLEMS

EVALUATION OF SELECTED CLASS III REQUIREMENTS
OF MIL-F-8785B (ASG), "FLYING QUALITIES OF PILOTED AIRPLANES"

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ABSTRACT

In 1969, the Air Force Flight Dynamics Laboratory issued a complete revision to the Military Specification, "Flying Qualities of Piloted Airplanes." This specification, MIL-F-8785B(ASG) has not yet been applied to the design or test of an operational, heavy-weight, cargo aircraft. In 1975, the Lockheed-Georgia Company completed a study program for the Flight Dynamics Laboratory in which C-5A flight test results were used to validate the requirements of MIL-F-8785B(ASG). The results of this study show that there are seven areas where the requirements of MIL-F-8785B(ASG) appear to be too stringent. Since completion of the C-5A program, the Lockheed-Georgia Company Flight Test Division instigated a study effort to compare selected sections of MIL-F-8785B(ASG) with flight test results from the C-5A, L-1011, C-141A, and YC-141B (Stretch C-141A).

INTRODUCTION

Results of the C-5A study effort presented, in Reference 1, indicate that the requirements of MIL-F-8785B(ASG) are too stringent for Class III airplanes in the following seven sections:

3.2.1.2 Phugoid Stability
3.2.2.1 Short Period Response
3.2.2.2 Control Forces in Maneuvering Flight
3.3.1.1 Lateral Directional Oscillations
3.3.1.2 Roll Mode (τ_R)
3.3.2.4 Sideslip Excursions
3.3.4 Roll Control Effectiveness

This paper presents a comparison of the requirements for the above sections of MIL-F-8785B(ASG) with flight test results from the C-5A, L-1011, C-141A, and YC-141B.

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AIRPLANE DESCRIPTION

C-5A

The C-5A is a long-range, all weather, high-altitude, high-subsonic, swept-wing, T-tailed airplane designed for use as a heavy logistic transport with relatively short-field takeoff and landing capability. The airplane is designed to airlift a wide variety of combat support equipment and personnel at payloads of up to 265,000 pounds. Aircraft gross weight ranges from 319,809 pounds empty to 769,000 pounds maximum design weight. Initial cruise altitude is 30,000 feet with cruise speeds of up to 470 knots true airspeed. It is powered by four General Electric TF-39 turbofan engines equipped with thrust reversers. Inflight reverse thrust is applied to the inboard engines for rapid or emergency descent. A retractable, high-flotation landing gear consisting of four six-wheel, bogie-type, main landing gears and a four-wheel, steerable nose gear enables the airplane to operate from paved or unpaved runways. The landing gear can be set at "crabbed" positions for takeoffs and landings in crosswinds. Some of the other unique design features of the airplane are a forward and aft cargo door system which enables straight-through loading and unloading, and a landing gear kneeling system. The kneeling system permits the cargo deck to be tilted nose-down or tail-down, or to be lowered in the level position. Aerial delivery of payloads through the aft cargo door is possible. Up to 200,000 pounds of payload may be dropped in multiple packages, and a single package of 86,000 pounds has been dropped in demonstration tests. Two auxiliary power units, one located in each main landing gear pod, are provided to supply electrical, pneumatic, and hydraulic power (through use of air turbine motors) for engine starting and for ground operation and maintenance requirements. Figure 1 presents a three-view drawing of the basic airplane.

Primary flight controls include ailerons, spoilers, rudders, and elevators. All surface hinge moments are provided by hydraulically powered actuators, and pilot "feel" is artificial. Control wheels, columns, and rudder pedals provide pilot or copilot inputs to the control valves through the mechanical linkage and cable systems. Hydraulic power is provided by four independent systems. Secondary flight controls include ground spoilers, leading-edge slats, pitch trim, and trailing-edge flaps.

Pitch and yaw/lateral SAS (Stability Augmentation Subsystem) are provided. Pitch SAS provides short-period pitch damping, Yaw/lateral SAS provides yaw damping, turn coordination, and spiral divergency control. The C-5 SAS is triple redundant, fail safe/fail operational. The actuator inputs are added in series with pilot inputs to control the surface actuators. The aircraft can be flown safely without SAS.

C-141A/YC-141B

The C-141A Starlifter is a long-range, high-subsonic, high-altitude, swept-wing, T-tailed airplane designed for use as a heavy logistic transport. The airplane is designed to airlift cargo or military personnel at payloads up to 70,000 pounds. Operating gross weight ranges from 130,000 pounds to 318,000 pounds maximum ramp weight. Design landing gross weight is 257,500 pounds with a maximum load factor of 2.5 g's.

The aircraft is powered by four Pratt & Whitney JT3D-5A (TF33-P-7 Military) turbofan engines which have twin-spool, axial-flow compressors. The engines, which are flat-rated at 21,000 pounds of thrust, are mounted individually in nacelles suspended below the wings. Each engine is equipped with target-type thrust reverser doors which are used only for a ground deceleration.

The aircraft is equipped with a fully retractable tricycle landing gear. The landing gear consists of two "four-wheel" bogie-type main gears which mount dual wheels forward and aft of the shock strut (in pods on each side of the aircraft) and a steerable, dual nose wheel. Anti-skid braking protection is installed on all main landing gear wheels.

Primary flight controls include ailerons, rudder, and elevator. All surface hinge moments are provided by hydraulically powered actuators, and pilot "feel" is artifical. Hydraulic power is provided by four independent systems. Secondary flight controls include flight/ground spoilers, pitch trim, and trailing-edge flaps. Stability augmentation consists of a yaw damper, but the airplane can be flown without the yaw damper operating. A "Q" feel system is incorporated in the longitudinal control system to provide positive speed stability.

Figure 2 presents a three-view drawing of the C-141A.

The YC-141B is similar to the C-141A with the following major modifications:

- o The fuselage length is increased 280 inches by adding a 160-inch plug forward of the wing and a 120-inch plug aft of the wing.
- o An aerial refueling receiver system using the Universal Aerial Refueling Receptacle Slipway Installation (UARRSI) is installed atop the fuselage in an aerodynamic fairing.

L-1011-100

The L-1011-100 is designed for transcontinental as well as short- and medium-range operation to handle high-density traffic markets. Primary features are the large spacious fuselage and application of the advanced technology high-bypass-ratio engines with low fuel consumption and low community noise. The general arrangement of the L-1011-100 with three Rolls-Royce RB-211-22-B engines is shown in Figure 3. The L-1011-100 is capable of operations at ranges up to 3735 nautical miles with a payload of 57,700 pounds at a normal cruise speed of Mach 0.85. Maximum takeoff gross weight is 466,000 pounds with an operational empty weight of 246,200 pounds. Maximum load factor in the cruise configuration is 2.5g's. Primary flight controls consist of an all movable stabilizer and geared elevator, a rudder, inboard and outboard ailerons, and spoilers for deceleration and roll control. The spoilers are also used for direct lift control for flaps-down operation. The pilot's control forces in pitch, yaw, and roll are artifically supplied by feel springs. Secondary flight controls include ground spoilers, leading-edge slats, pitch trim, trailing-edge flaps, directional trim and lateral trim. The

stability augmentation consist of a yaw damper and a Mach trim compensator.

DISCUSSION OF TEST RESULTS

The C-5A and C-141A stability and control data used in this paper were obtained during combined Lockheed and Air Force Category I/II test programs. The C-5A test results are presented in Reference 2, and the C-141A data are presented in Reference 3. The YC-141B data discussed herein were obtained during the recently completed Lockheed/Air Force YC-141B Development Flight Test Program, and these data are presented in Reference 4. The L-1011 results were computed from basic aerodynamic data as updated from the Flight Test Certification Program.

The scope of this paper does not permit the inclusion of all applicable airplane configurations and failure states relative to the seven sections listed in the Introduction. In most instances only cruise configuration (Category B) data are discussed. However, where convenient, landing and approach configuration (Category C) data are also included. Additionally, all data are applicable to Level 1 conditions except the data for the lateral-directional damping section. These data were obtained with the yaw damper inoperative, and it is assumed that the results correspond to Level 2 requirements.

Phugoid Stability

Results from C-5A, C-141A, YC-141B, and L-1011 phugoid stability tests are presented in Figure 4 in terms of damping ratio versus undamped natural frequency of the phugoid oscillation. The MIL-F-8785B(ASG) Level 1 requirement, which states that ζp shall be at least 0.04, is also presented for comparison. The C-5A, C-141A, and L-1011 data are applicable for both forward and aft center of gravity positions; the YC-141B data are for aft c.g. only. The data presented in Figure 4 show that the C-5A and L-1011 do not comply with the minimum damping ratio requirement of 0.04. Pilot comments for the C-5A corresponding to these results average about 3.5 which are for Level 1 conditions. General pilot comments for the L-1011 indicate values similar to the C-5A.

For the Level 2 test conditions, the damping ratio is permitted to drop to zero with a corresponding degradation in pilot rating. Here the inconsistency appears to exist in the specification requirements for a Class III airplane. Test results correspond to Level 2 requirements, but the pilot ratings correspond to Level 1 requirements. This inconsistency is considered to exist because of the period ω_{np} has not been taken into consideration. The C-5A results which fell below the 0.04 damping requirement had a period of at least one minute, which probably affected pilot ratings considerably. Therefore, it is concluded that the 0.04 damping requirement for Level 1 should be relaxed provided that the frequency of the oscillation is low enough not to affect trimmability or longitudinal control. It is also evident from these data that the application of MIL-F-8785B(ASG) longitudinal phugoid requirements to the C-5A initial design, in lieu of MIL-F-8785(ASG) requirements, would have had an insignificant effect on overall pilot ratings but would have had a significant effect on the initial design and resulting cost.

Short-Period Response

Short-period response characteristics for the four airplanes are shown in Figure 5 in terms of the undamped natural frequency of the short-period oscillation versus the normal acceleration change per unit change in angle of attack, along with the specification requirements for the Category B flight phase. These data show that the C-141A and YC-141B comply with the Level 1 requirements but that the C-5A and L-1011 do not.

Based on the C5-A and L-1011 short-period data presented herein, the Level 1 and Level 2 frequency requirement envelopes appear to be too high for all Flight Phase categories for a Class III airplane. Pilot comments indicate that the short-period response for the C-5A and L-1011 correspond to Level 1 conditions. However, test results do not completely agree with specification requirements. Although the terrain-following flight phase is not yet used on C-5A fleet aircraft, the in-flight refueling phase has been used with very satisfactory results. Therefore, it appears that, for Flight Phase B, the lower frequency limit should be reduced for altitudes above approximately 20,000 feet.

Elevator Control Force Gradient

Figure 6 summarizes elevator control force gradient characteristics for the C-5A, C-141A, YC-141B, and the L-1011 for category B at forward and aft center of gravity conditions. This summary shows that the L-1011 complies with the maximum and minimum control force gradient requirements but that the C-5A and C-141A/YC-141B do not. The C-5A at forward c.g. compares favorably with the Level 1 maximum values; however, the gradients at aft c.g. fall below the Level 1 and Level 2 boundaries. The C-141A/YC-141B data slightly exceed the level maximum limit at forward c.g. Pilot comments for the C-141A/YC-141B support the maximum boundary. However, C-5A comments do not support the aft c.g. minimum boundary. The minimum boundary for Level 1 requirements appears to be too high.

Lateral Directional Damping

Lateral directional damping characteristics for the C-5A, C-141A, YC-141B, and L-1011 are summarized in Figure 7 for the Category B Flight Phase with the yaw damper inoperative. Each of these airplanes comply with the minimum damping (0.08) and frequency (0.40) requirements for Level 1, but as Figure 7 shows, the Level 2 requirement in terms of the minimum produce ($\omega_{nd} \zeta_d$) values of 0.05 is not met by the C-5A or the C-141A and YC-141B. Relative to the minimum damping and frequency requirements of 0.02 and 0.40, respectively, the C-5A results show satisfactory compliance. The C-141A and C-141B data comply with the minimum frequency requirements, but the minimum damping requirement is not met.

An evaluation of the C-141A dutch-roll recovery techniques with the yaw damper inoperative was conducted by the Air Force Flight Test Center in February 1977. Results of the tests, presented in Reference 5, show Harper-Cooper rating values ranging from 2.0 to 5.0, using aileron only for recovery, which is the recommended Flight Handbook procedure. Over 100 dutch-roll maneuvers were accomplished during the evaluation, which

consisted of regaining control of the aircraft and returning to a wings-level attitude from bank angles as high as ± 45 degrees. It should also be noted that evaluating pilots do not rate operation of the C-5A with the stability augmentation system off below the suggested Level 2 guidelines (6.5 Harper-Cooper rating scale). These data strongly indicate that the Level 2 minimum $\zeta_d \, \omega_{nd}$ requirement of 0.05 rad/sec is too stringent.

Roll Mode Time Constant

Figure 8 summarizes roll mode time constant values for the C-5A, C-141A, YC-141B, and L-1011 for Level 1 and Category B Flight Phase. The C-141A and YC-141B data comply with the Level 1 roll-mode time constant requirement of 1.4. However, the C-5A and the L-1011 data show values consistently in excess of the 1.4 requirement.

One of the significant characteristics following the input of rapid full lateral control on the C-5A is that the initial rolling acceleration produces a very noticeable "side kick" or lateral acceleration component in the cockpit and in the troop compartment, since the cockpit and troop compartment are located considerably above the principal roll axis of the airplane. For normal operation, this characteristic can be avoided by initially using slow lateral contol input and then increasing the rate of input until the desired airplane response is obtained. In situations which require abrupt full control input, this characteristic will be noticed; however, it will not unduly restrict the use of full control when required.

The purpose of the roll-mode requirement is to describe the shape of the roll rate trace which is essentially defining the average rolling acceleration. The C-5A does not meet the Level 1 requirements. To achieve the Level 1 roll-mode time constant on the C-5A would produce an even more objectionable condition. This problem should be recognized in the roll-mode time constant requirement.

Sideslip Excursions

Sideslip excursion data for the C-5A, C-141A, YC-141B, and L-1011 are presented in Figure 9 in the form of the ratio of sideslip increment, to the parameter K versus calibrated airspeed. One-half of the dutch roll period has been used to obtain the $\Delta\beta$ parameter, since the dutch roll period for each airplane varies from approximately 6 seconds to 11 seconds. This criterion is specified in MIL-F-8785B(ASG). As the data show, neither airplane completely complies with the requirements. The sideslip excursions for the C-5A, C-141A/YC-141B and the L-1011 are not considered excessive. Additionally, the high sideslip excursion angles shown in Figure 9 for the C-5A, the C-141A/YC-141B, and the L-1011 are not considered to be consistent with normal operation due to requirement to hold the aileron command fixed until the bank angle has changed at least 90 degrees. Pilot rating data obtained during the YC-141B flight test program show a value of 2 (Harper-Cooper Rating Scale) with augmentation operative and 4 with the augmentation inoperative. These data indicate that the handling characteristics correspond to Level 1 conditions even though the data fall outside Level 1 requirements at the lower airspeeds.

It should be noted that the L-1011 nearly complies with roll performance requirements, but the sideslip excursions created as a result, as shown herein, exceed allowable limits. The L-1011 sideslip excursion have not prompted any objectionable comments from flight test or airline pilots.

Roll Performance

C-5A, C-141A, YC-141B and L-1011 Category B test results are presented in Figure 10, along with applicable MIL-F-8785B(ASG) requirements. These data show that only the L-1011 rolling performance compare favorably with the Level 1 requirements.

Although neither the C-5A nor the C-141A/YC-141B comply with the rolling performance requirements, qualitative pilot comments indicate that both airplanes have acceptable rolling performance in the cruise configuration. Results obtained during the YC-141B flight test program show a Harper-Cooper rating of 2.0 with augmentation on and 4.0 with the augmentation off for the cruise configuration. L-1011 rolling performance very nearly meets the requirements; however, abnormal sideslip angles are generated due to spoiler drag when full lateral control is used for an extended period of time.

For the C-5A to meet the Level 1 requirements, the lateral control system would have to be improved to attain a higher bank-angle change in the first second of roll. On an aircraft with a very large rolling moment of inertia, this would be difficult to accomplish. Increasing the initial roll response of the C-5A would further aggrevate the very noticeable side kick, or lateral acceleration component, in the cockpit and troop compartment that is experienced during full abrupt control input. The side kick occurs since the cockpit and troop compartment are located considerably above the principal roll axis of the airplane.

SUMMARY AND CONCLUSION

In spite of the brevity of this paper, it is felt that sufficient data have been assembled from four Class III airplanes to show that significant differences exist between certain sections of MIL-F-8785B(ASG) and presently acceptable handling characteristics criteria. The most significant differences between the specification and the data are those sections dealing with lateral control: roll mode (τ_r), sideslip excursions, and roll control effectiveness. These sections are considered too stringent for Level 1 conditions. The data also indicate that some degree of adjustment in Level 1 requirement boundaries is warranted for the following sections of the specification:

Phugoid Stability
Short Period Response
Control Forces in Maneuvering Flight

The data strongly indicate that the section on lateral directional damping is too restrictive for Level 2 conditions. Specific recommendations for revising the aforementioned sections of MIL-F-8785B(ASG) are not made due to the lack of test results for the various failure states.

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3. McCabe, J. H. and Benefield, T. D., "C-141A Category II Stability and Control Test," FTC-TR-66-5, June 1966.

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5. Picha, D. G. and Klein, H., "C-141A Dutch Roll Recovery," AFFTC Technical Letter Report, February 1977.

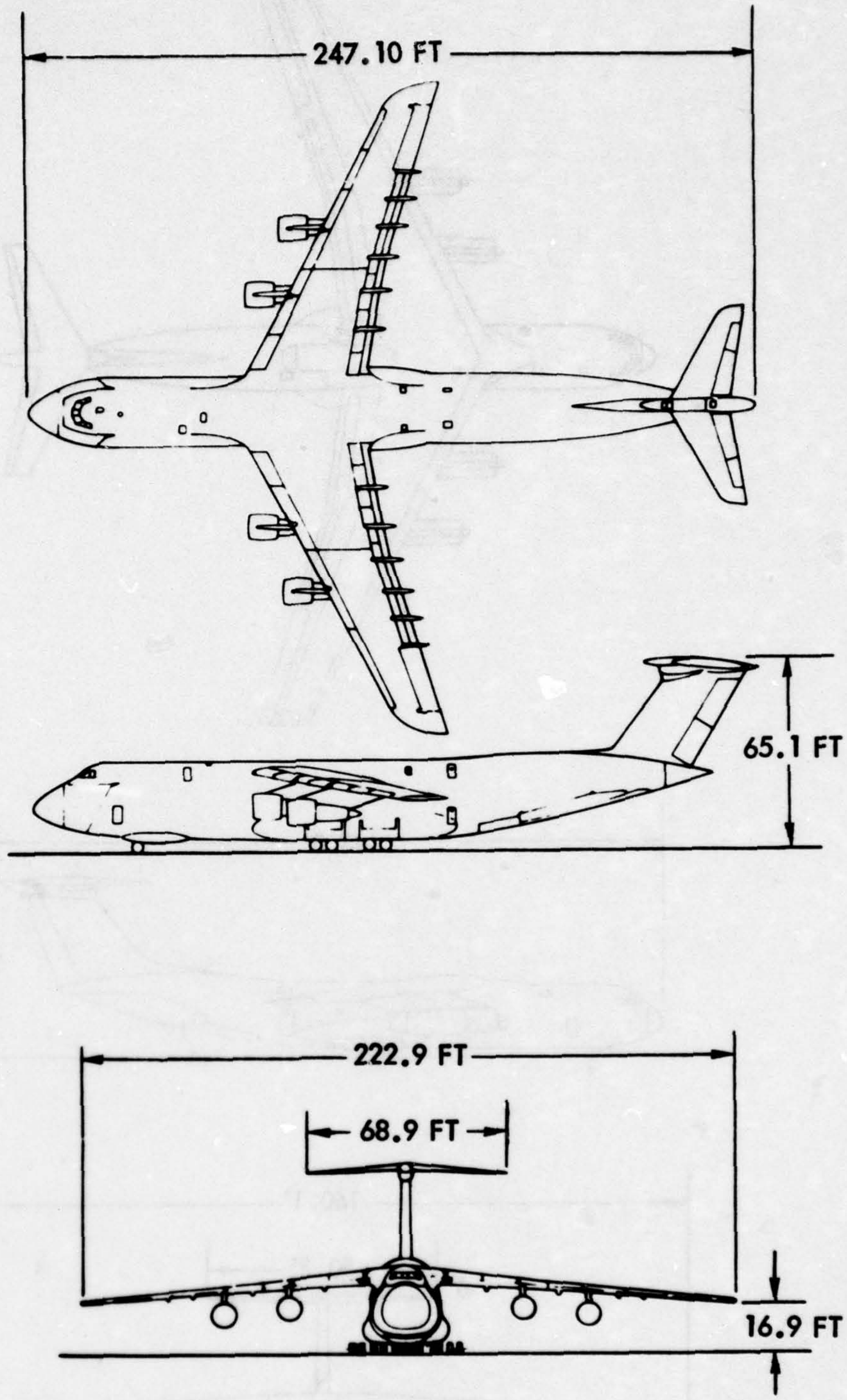

Figure 1. C-5A General Arrangement

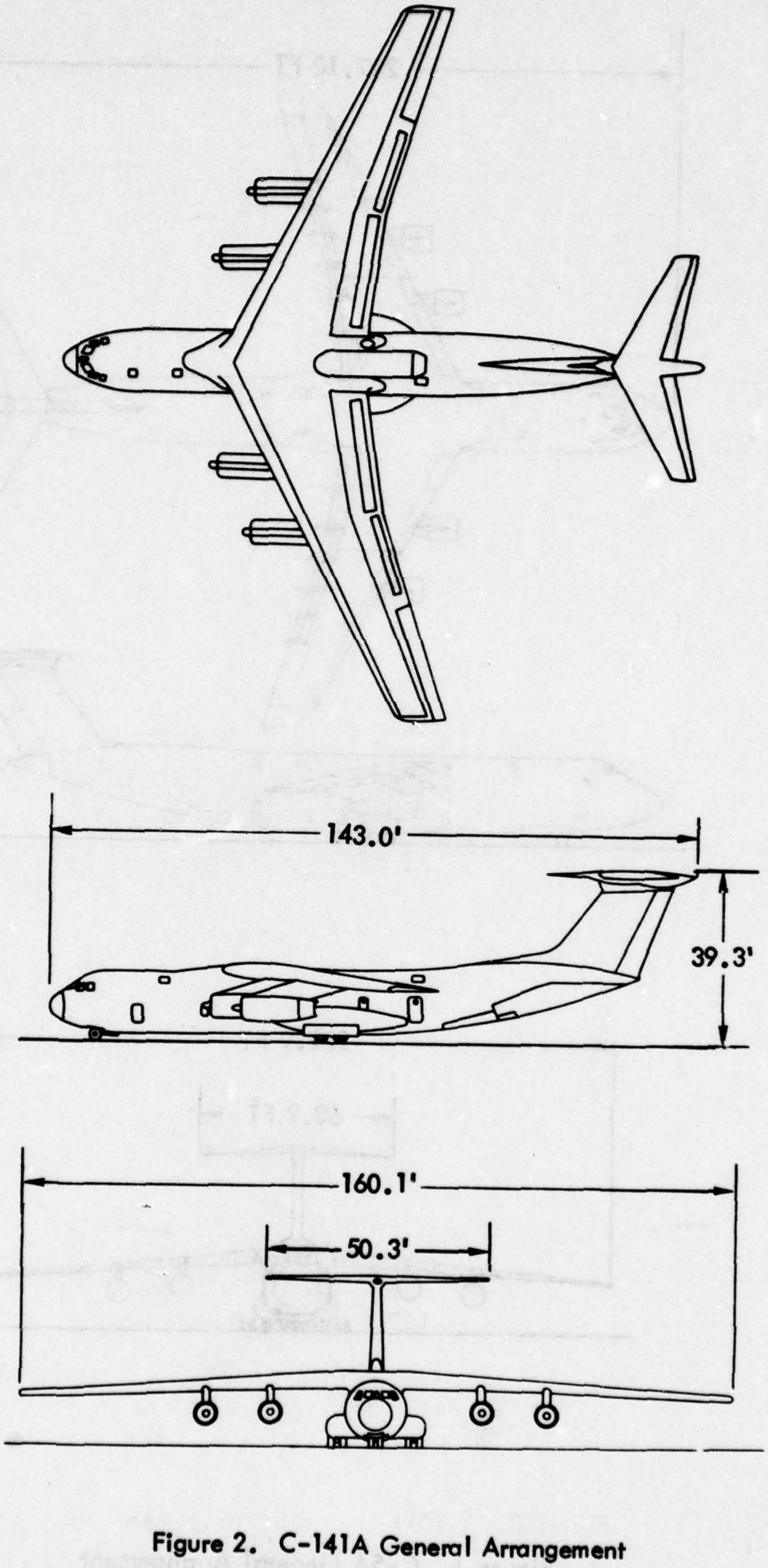

Figure 2. C-141A General Arrangement

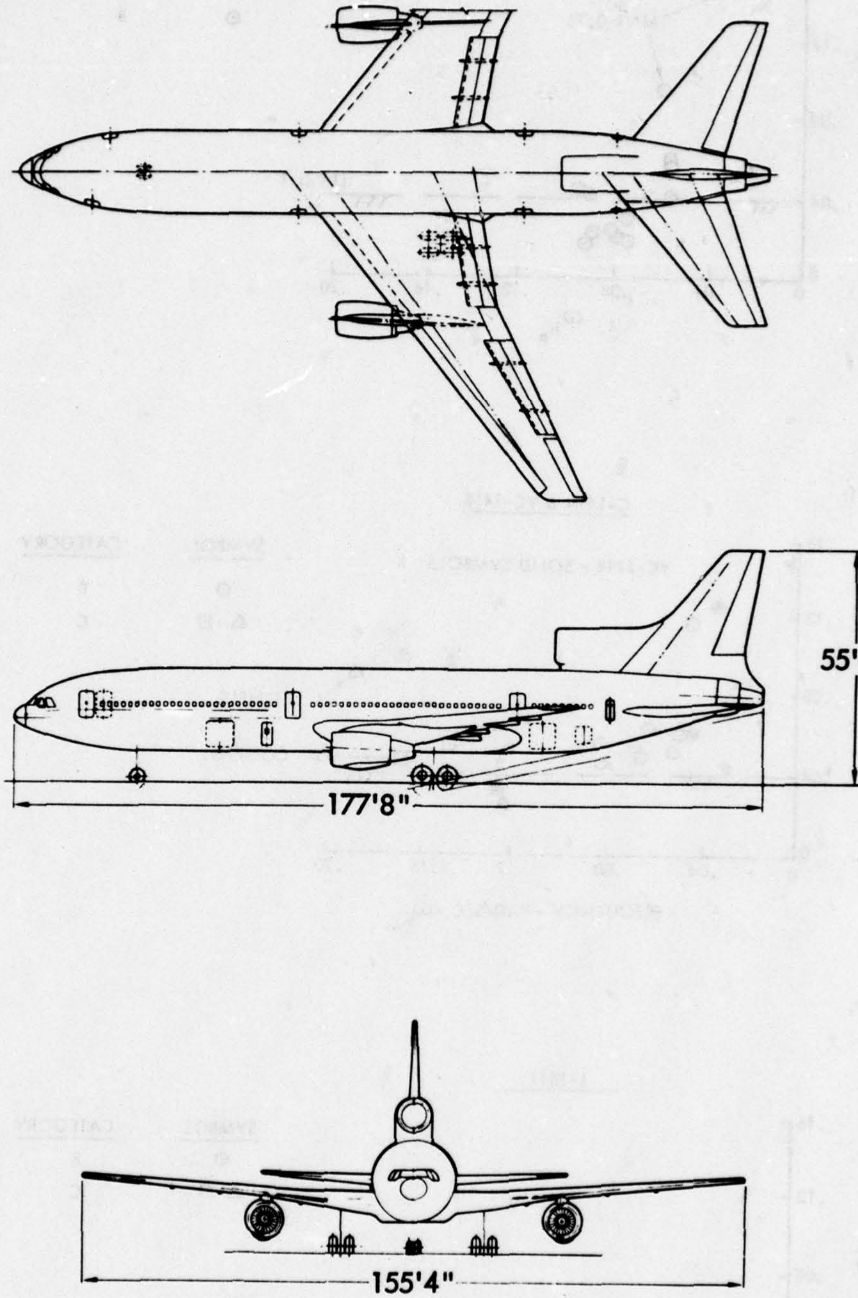

Figure 3. L-1011 General Arrangement

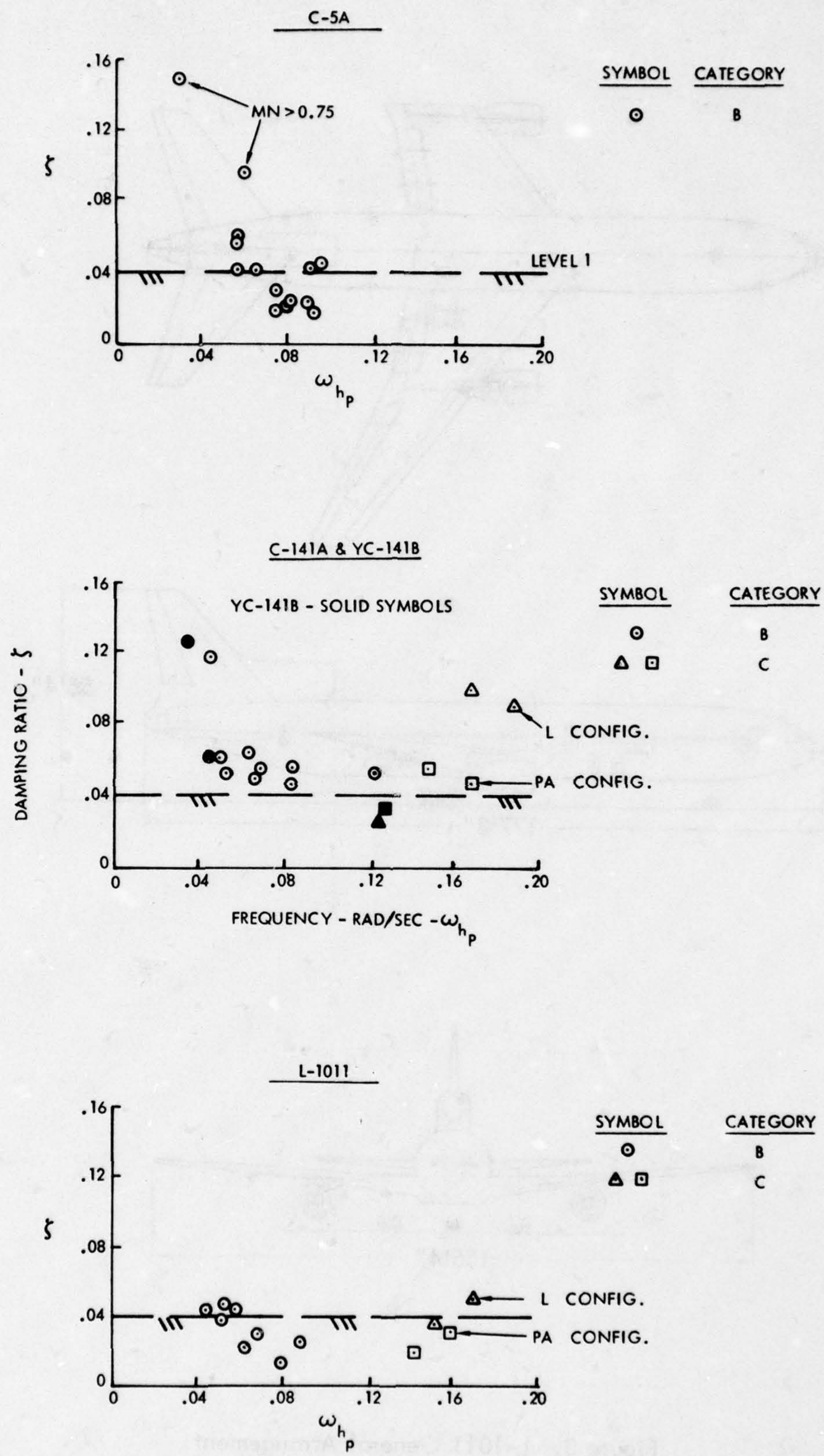

Figure 4. Phugoid Stability

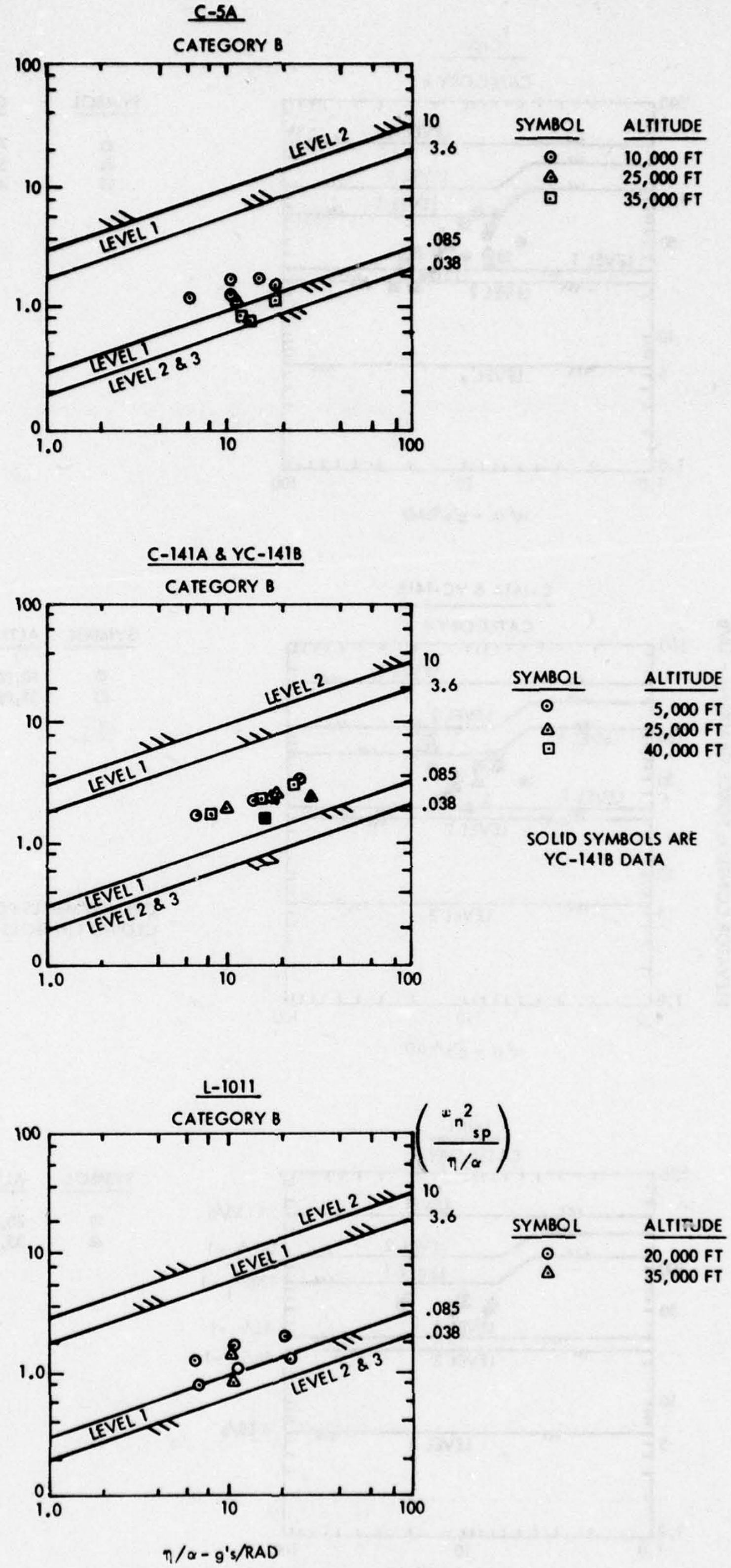

Figure 5. Short Period Response

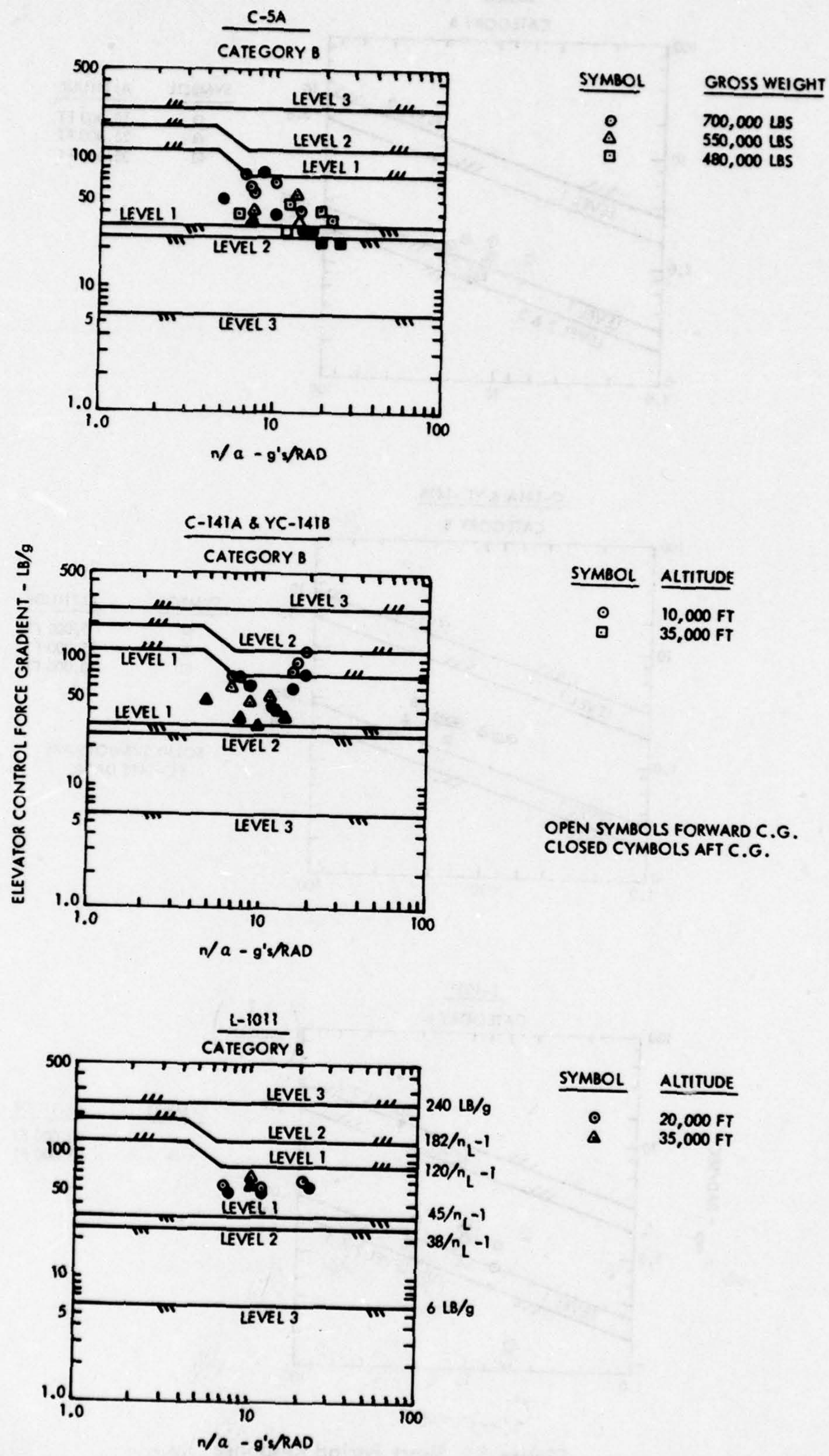

Figure 6. Elevator Control Force Gradient

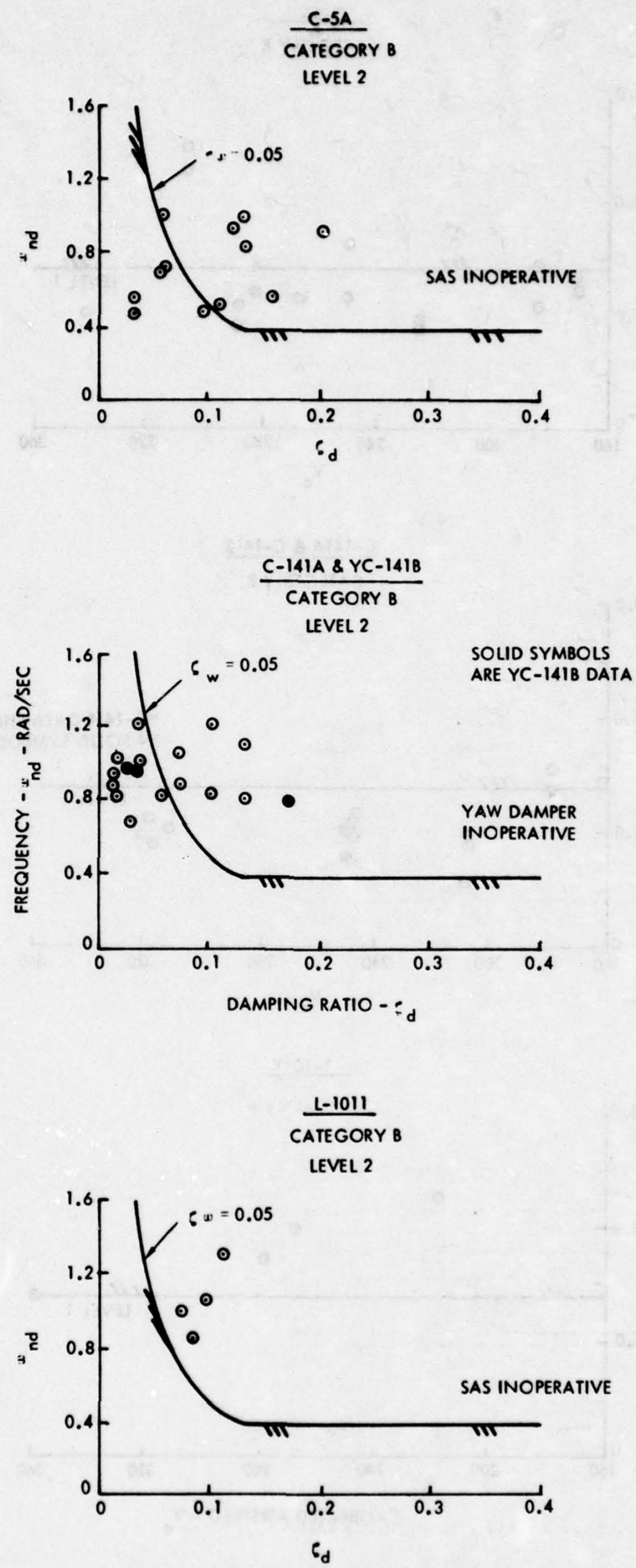

Figure 7. Lateral Directional Damping

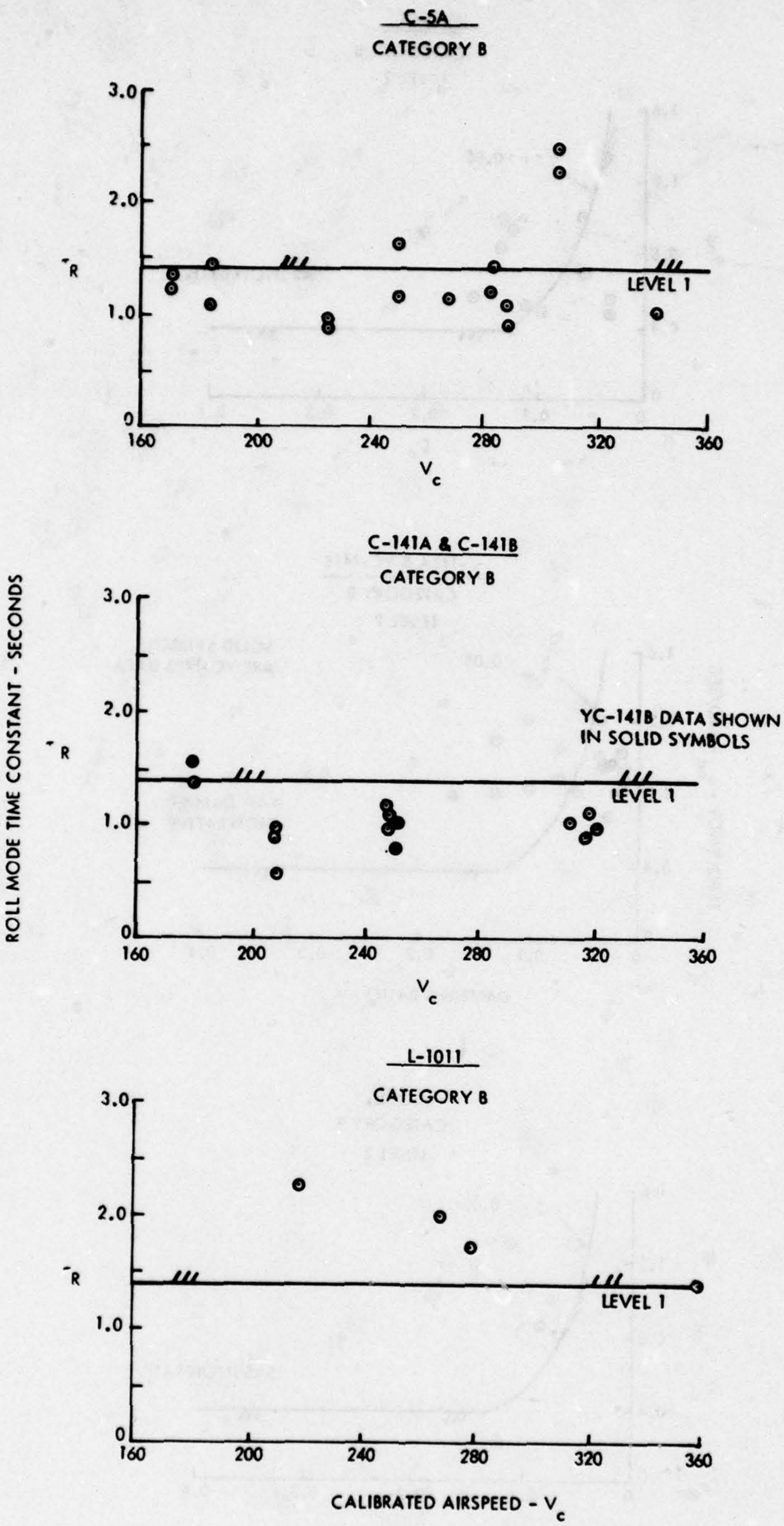

Figure 8. Roll Mode Time Constant

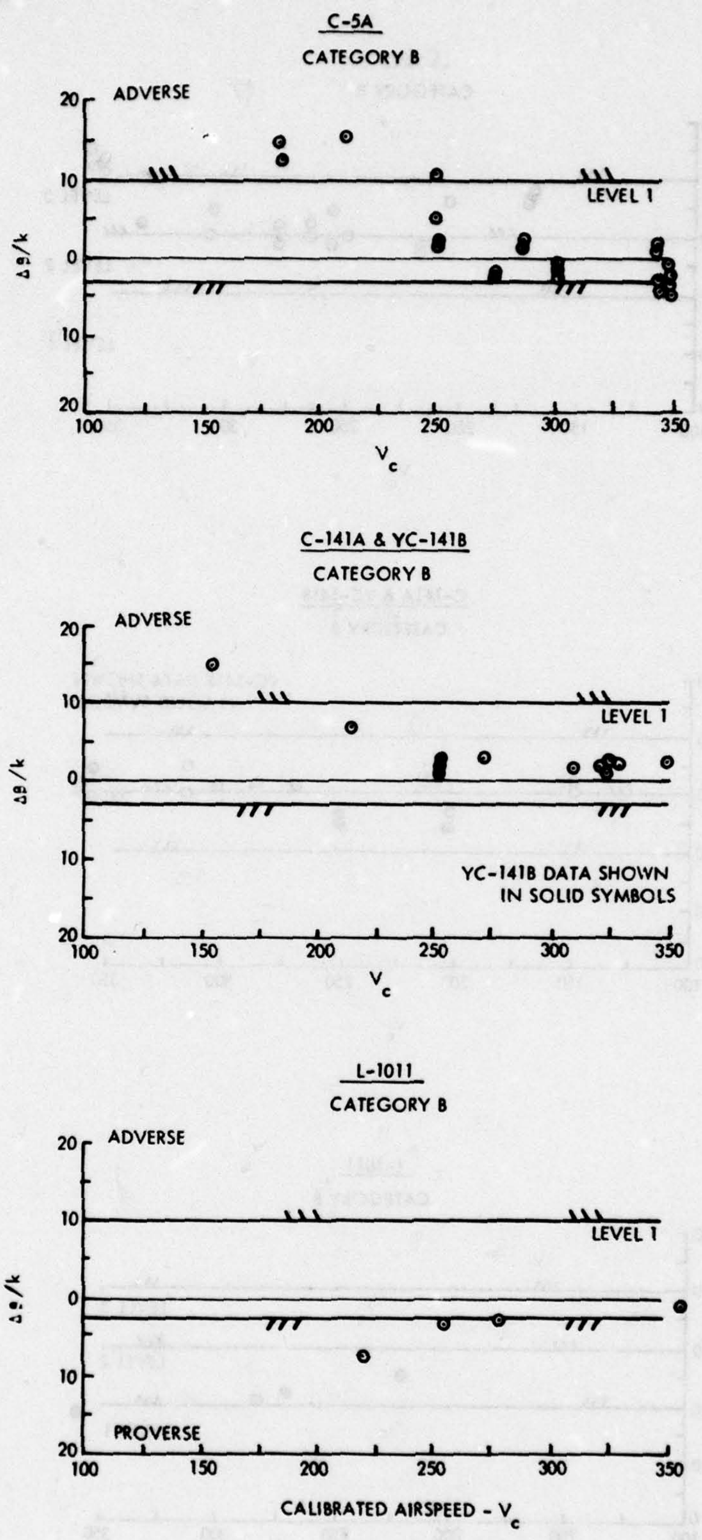

Figure 9. Sideslip Excursions

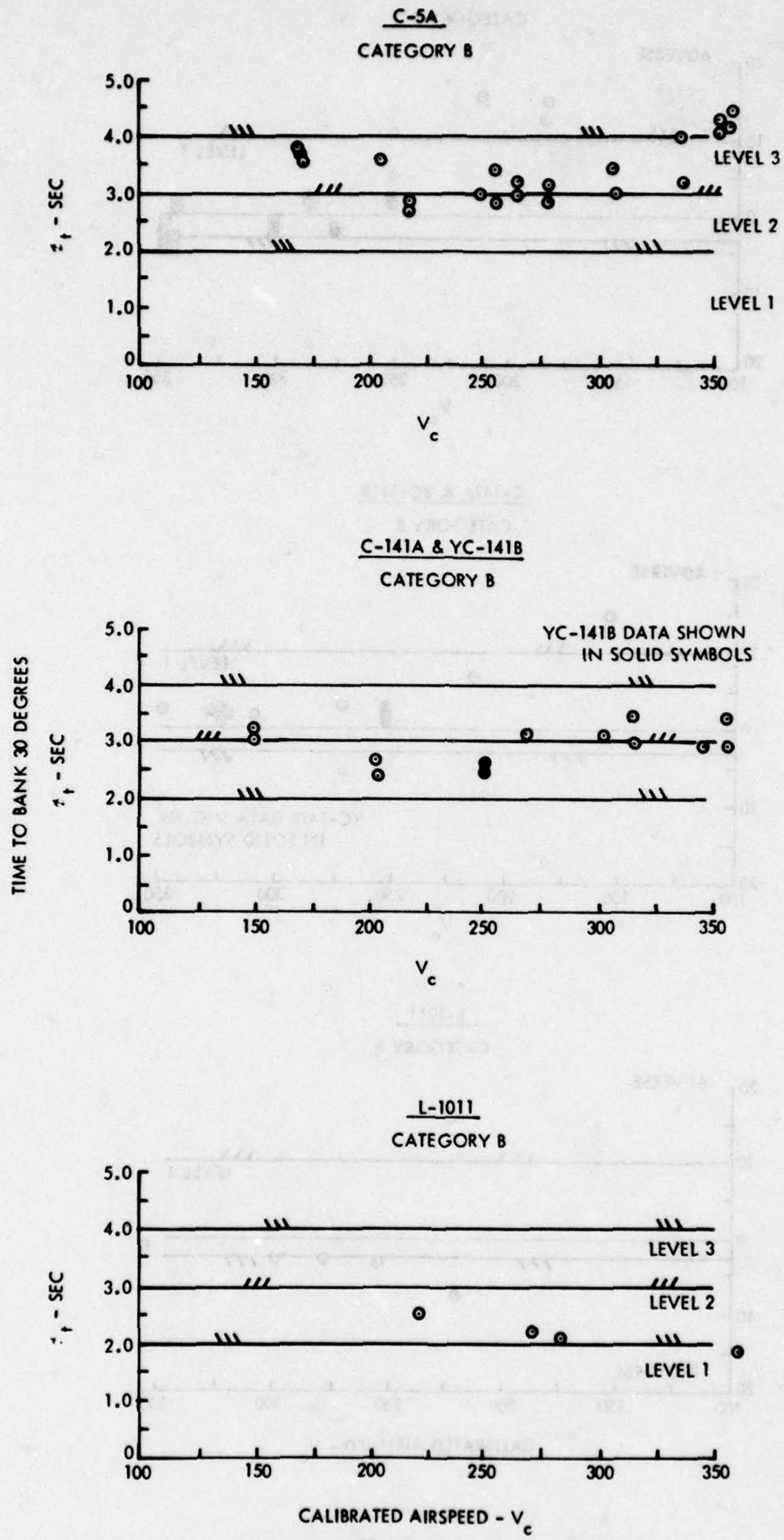

Figure 10. Roll Performance

Bill Rickard, Douglas Aircraft: Douglas feels the ±60° wheel throw is too small. What is Lockheed's position?
Answer: Lockheed has a proposal in to Warner-Robins to reduce the C-141B wheel throw to ±70°. We feel that 110° wheel throw is too large for Class III aircraft.

Comment by Frank Wilson, Lockheed-Georgia on Cliff Wither's paper: None of the four aircraft discussed (C-5A, C-141A, C-141B, or L-1011) were designed to meet requirements of MIL-F-8785B(ASG). The C-5A and C-141A,B were designed to meet MIL-F-8785(ASG).

Bill Rickard, Douglas Aircraft: What is the wheel throw for the L-1011?
Carl Anderson, Lockheed-California: It is ±85°.

SECTION IV

SPECIAL PROBLEMS 1

TASK-ORIENTED FLYING QUALITIES FOR AIR-TO-GROUND GUN ATTACK

June 1978

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TASK - ORIENTED FLYING QUALITIES FOR AIR-TO-GROUND GUN ATTACK

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Abstract

The A-10 Stability Augmentation System (SAS), designed to provide good overall flying qualities based on the standard MIL-F-8785B criteria, performed very well during prototype testing and early operational deployment. As user experience was gained, progressively more aggressive close air support tactics were developed. It was recognized that the SAS could benefit from reevaluation, with a view toward determining potential low cost improvements in performance. The starting point of the evolution of the task-oriented SAS was a study of current close air support weapon delivery maneuvers, which established two typical evaluation maneuver scenarios, namely: curvilinear strafe and abrupt heading change maneuvers. These maneuvers, in turn, motivated the definition of the task-oriented control variable, the cross-track component of the perceived hit point. The ability to rapidly and predictably produce changes in this quantity with 1-second aileron doublet inputs was chosen as the evaluation criterion. Examination of the contribution of lateral stick motions to cross-track hit point led to the definition of five candidate beta-dot ($\dot{\beta}$) systems for the SAS. The most cost-effective form of the SAS was evaluated via manned simulation and motivated a flight test program. Flight test results, showing tracking time reductions of more than 100 percent were in excellent agreement with the previous work. The A-10 aircraft will be equipped with this task-oriented SAS in the near future.

1. Introduction

In the close air support (CAS) role, a wide variety of attack maneuvers may be characterized by three general phases. The first phase, initiated by the pilot perceiving the target, consists of a target acquisition maneuver. This maneuver consists of a rapid rollin toward the target while a normal load factor of 4-5g is developed. The roll angle and load factor are maintained until the gun cross or pipper line of sight is near the target. At this point a rollout to wings level, together with a load factor reduction to 1g, occurs. The second phase of the attack maneuver is the weapon delivery or tracking/firing phase. In this phase, the errors present at the conclusion of the target acquisition phase must be eliminated, and the pipper should be maintained on the target while the gun is fired. The final portion of the attack is characterized by a break phase, which consists of a gross maneuver generally intended to place the aircraft in position for another attack while maximizing aircraft survival. Figure 1 presents a specific example of this ground attack maneuver scenario. Each of these attack phases will now be examined to obtain the functional requirements.

The target acquisition and break phases are considered first. Both of these phases involve gross maneuvers. A gross maneuver is defined as one utilized to produce a large change in the aircraft velocity vector. The target acquisition gross maneuver has an additional essential requirement, namely, that the terminal direction of the velocity vector must be highly predictable. The realization of the proper terminal orientation of the velocity vector is vital to minimizing the duration of the relatively vulnerable weapon delivery phase. The target

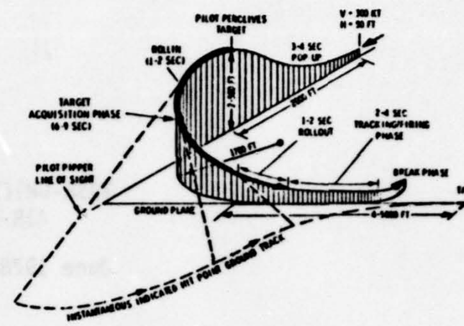

Figure 1. Ground Attack Maneuver Scenario

acquisition and break maneuvers are achieved (as is any gross maneuver) by developing a large, unbalanced, aerodynamic force vector (lift) and orienting it with roll angle control so that the resultant, with the gravitational force vector, is oriented in the direction of the desired velocity increment. This type of maneuver requires that large amounts of lift be obtained at optimum low altitude attack speeds (275-350 knots) and that the aircraft show excellent roll control characteristics in terms of speed of response, while maintaining adequate turn coordination when necessary.

The weapon delivery phase is typified by relatively straight, roughly 1g flight. There are, however, two distinct types of functional requirements. These requirements are motivated by unguided bombs as one weapon type and the GAU-8 gun as the other. Other weapons will have requirements encompassed by the requirements of these two weapons. The impact point of an unguided bomb is determined essentially by the aircraft velocity and position vectors at the time of weapon release. The aircraft position is, of course, a consequence of velocity over a time interval, and therefore, the fundamental requirement is for rapid and precise velocity vector control. Rounds from the GAU-8 gun, however, impact at points largely determined by the direction of the pipper line of sight when the round was fired. The gun, therefore, requires that the attitude of the aircraft (and consequently of the gun) be precisely controlled.

The A-10 SAS was originally designed to provide good overall flying qualities, as defined by MIL-F-8785B. During the original deployment and operational utilization of the aircraft, the augmented dynamic response was found to be excellent. As the aggressiveness of the ground attack maneuvers increased, it was determined that a task-oriented SAS was required in order to realize the inherent dynamic performance capabilities of the aircraft in this evolving maneuver environment. The purpose of this paper is to contrast the dynamic performance assessment obtained from application of MIL-F-8785B with the assessment derived from use of a task oriented performance measure. The original lateral-directional SAS is initially defined. The development of the task oriented SAS is then presented.

The response characteristics of the systems are then compared: first, in terms of application of MIL-F-8785B for response evolution, then, in terms of task oriented response. The results of simulation, and, finally, flight test are then presented.

2. Original SAS

The original lateral directional stability augmentation system is presented in Figure 2. This system was selected on the basis of extreme simplicity and capability to satisfy the requirements of the flying qualities specification. Figure 2 shows that the system consists of cancelled yaw rate feedback to provide adequate damping, together with an aileron rudder interconnect to inhibit any adverse yaw tendencies. This system provided level 1 lateral directional flying qualities during the prototype flight test phase and original operational deployment of the aircraft. As the evolving maneuver environment became more aggressive, it became apparent, however, that the response requirements of 8785B did not guarantee optimum air-to-ground gun attack flying qualities. A study was initiated to gain insight into these specific requirements.

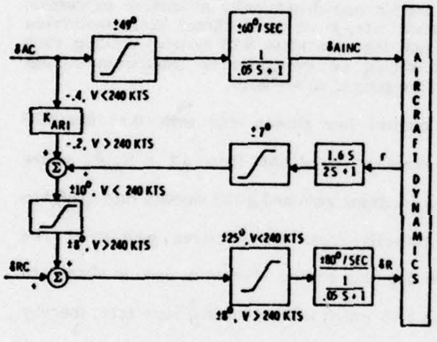

Figure 2. Original Lateral - Directional SAS

3. Kinematic Analysis

Analysis of gun camera film, pilot comments, etc., together with consideration of the functional requirements for CAS presented in the previous section, motivated the selection of indicated hit point steering as the quantity which the SAS should be synthesized to control in a smooth and rapid fashion.

In order to provide control of this vector, it is necessary to derive an analytical expression for it in terms of the aircraft position and orientation.

The required perceived hit point expressions were developed using the vectors and geometry defined in Figure 3. The results are:

$$R_{H_x} = x - z\left[\frac{\cos\theta\cos\psi + \sigma_c\ (\sin\phi\sin\psi + \cos\phi\sin\theta\cos\psi)}{-\sin\theta + \sigma_c\ \cos\phi\cos\theta}\right]$$

$$R_{H_y} = y - z\left[\frac{\cos\theta\sin\psi + \sigma_c\ (\cos\phi\sin\theta\sin\psi - \sin\phi\cos\psi)}{-\sin\theta + \sigma_c\ \cos\phi\cos\theta}\right]$$

Where: x, y, z are the position coordinates in the R-Frame of Figure 3, ψ, θ, and ϕ are the conventional Euler angles, and σ_c is the gun sight depression angle. These expressions will be used as the basis for the task orientated SAS synthesis.

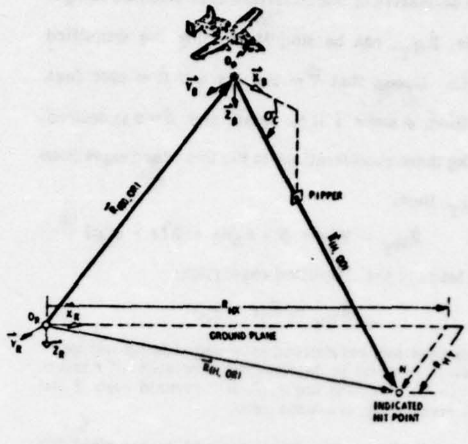

Figure 3. Fixed Reticle Air-to-Ground Weapon Delivery Kinematics

4. Simplified Scenario

The complete kinematic and dynamic analysis of the gunnery scenario presented in the introduction represents a formidable analytical task, and requires simplification in order to gain insight into the problem. Therefore, attention is focused on that portion of the maneuver during which it is most critical to have good control of hit point (pipper) motion, viz. the final portion of the rollout and the track phase.

The simplification shall consider the aircraft at an average slant range of about 4000 feet, velocity of 300 knots, (500 fps), and shall restrict the vehicle motion to a 10-degree inclined plane. This is typical of a low-dive angle rollout maneuver.

Making the usual perturbation analysis assumptions, the linearized expression for the cross track component of the perceived hit point rate can be written as:

$$\dot{R}_{H_y} = V(\psi + \beta - \alpha_0\phi) + \bar{R}(r - \sigma_c p)$$

Where: V is the aircraft velocity, $\bar{R}$ is the average range, α_0 is the trim angle of attack, and β is the sideslip angle.

5. System Synthesis

This section presents the reasoning which led to the control law selection. It was recognized at the outset that control of the perceived hit point (pipper line of sight) should be accomplished in such a way that minimum sideslip angle be maintained, since significant sideslip could cause discrepancies between actual and perceived hit points. The aileron was chosen to be the primary control, since "feet-on-the-floor" maneuvering is desirable. If the target appears through the Head Up Display (HUD) to the right of the pipper, a natural pilot response is to roll to the right. It is desirable to have the gun cross move to the right without any pendulum effect (initial non-minimum phase motion; i.e., motion of the gun cross initially opposite to the desired direction). In order that the perceived hit point move to the right, it is necessary that $\dot{R}_{H_Y} > 0$, whenever $p > 0$.

The derivative of the linearized task oriented control variable, $\dot{R}_{H_Y}$, can be simplified using the simplified scenario. Recall that $V \approx 500$ fps and $\bar{R} \approx 4000$ feet. In addition, ψ and ϕ will be small, and $\beta \approx 0$ is desired. Applying these considerations to the linearized expression for R_{H_Y}, then:

$$\dot{R}_{H_Y} = V(\dot{\psi} + \dot{\beta} - \alpha_0\dot{\phi}) + \bar{R}(r - \sigma_c p)$$

which leads to the simplified expression:

$$\dot{R}_{H_Y} \approx \bar{R}(r - \sigma_c p)$$

This equation was derived assuming sideslip was zero. It is now of interest to determine the relation of r and p, when $\beta = 0$. Sideslip angle, β, will remain zero if the time derivative, $\dot{\beta}$, is always zero.

The aircraft side acceleration equation can yield this relation. The equation is:

$$\dot{v} + ur - wp = g \sin\phi \cos\theta + Y_A/m$$

Where Y_A = aerodynamic side force, and u, v, and w are aircraft axial, side, and vertical velocities. Using $u \approx V$, $\alpha = w/V$, $\beta = v/V$, dividing through by velocity, and rearranging terms yields:

$$\dot{\beta} = -r + \alpha p + (g \sin\phi \cos\theta/V) + Y_A/mV$$

If $\dot{\beta} = 0$, then:

$$r = \alpha p + (g \sin\phi \cos\theta/V) + Y_A/mV$$

Assuming Y_A/mV is negligible, then:

$$r = \alpha p + g \sin\phi \cos\theta/V$$

For ϕ small, g $\sin\phi/V$ is negligible, and $r \approx \alpha p$.

Substituting this approximation for r into the simplified expression for $\dot{R}_{H_Y}$, then:

$$\dot{R}_{H_Y} \approx \bar{R}(\alpha p - \sigma_c p) = \bar{R}p(\alpha - \sigma_c)$$

This indicates that whenever the angle of attack is the same as the gun depression angle, the perceived hit point will not move in the $\hat{Y}_R$ direction, in response to roll rate. In other words, the aircraft would roll about the pipper and the pilot could not make a correction via ailerons. This situation would be desirable if the pilot were able to roll out on the target with no error. A more usual situation is that some initial errors exist, and it is not desirable to roll about the gun line. It is more desirable to roll about an axis which is below the gun line so that a proper initial motion of the perceived hit point (gun line) is generated. In order to accomplish this, a bias is introduced into the $\dot{\beta}$ equation, i.e., if it is required that $\dot{R}_{H_Y} > 0$ when $p > o$ for all $\alpha > 0$, then it is appropriate to replace the α term in the $\dot{\beta}$ equation by $\alpha + \alpha_\beta$ where $\alpha_\beta = \sigma_c$. This results in:

$$\dot{R}_{H_Y} = \bar{R}\alpha p$$

So far as control of R_{H_Y} is by the use of aileron control for this simplified scenario, the result can be derived based on the one degree-of-freedom (1 DOF). roll rate response as:

$$\dot{R}_{H_Y}(s)/\delta_A = \bar{R}\alpha p(s)/\delta A$$

For the design flight condition:

$$\dot{R}_{H_Y}(s)/\delta_A = 3628/(s + 5.23)$$

This simple and dynamically attractive expression for hit point rate, in terms of aileron input, motivated the decision that candidate SAS systems yielding such a result should be evaluated by simulation studies based on the simplified scenario.

The control law chosen was such that the SAS rudder command is of the form $\delta R = K_{\dot{\beta}}\hat{\dot{\beta}}$, where $K_{\dot{\beta}}$ is the feedback gain and $\hat{\dot{\beta}}$ the sideslip rate estimate derived directly from the sideforce equation. The rationale for this choice of control law is simply to provide a SAS which will maintain $\hat{\dot{\beta}}$ near zero, thereby yielding a dynamic relationship between $\dot{R}_{H_Y}$ and δ_A, which will approximate the favorable one derived above via linearized analysis.

6. Flying Qualities Comparison

The purpose of this section is to provide a brief outline of the differences in flying qualities between the original SAS and the task oriented SAS as obtained from the flying qualities specification. The response of both systems to a maximum roll input is presented in Figure 4. In terms of the response requirements of paragraph 3.3.2.4 Sideslip Excursions of MIL-F-8785B the response of both systems in excellent, i.e., level 1. The results of application of paragraph 3.3.2.4.1 Additional sideslip requirement for small inputs are presented in Figure 5. The results of this Figure also shows both SAS responses to be excellent. These results show that a more discriminating criterion is required for the gun attack problem. The next section describes the selected criterion as well as initial simulation results.

7. Preliminary Simulator Studies

A three degree-of-freedom (3 DOF) lateral/directional flight simulation was set up at Fairchild Republic for extensive analysis of the ground attack mode. The objectives of this simulation were to:

- **Validate response criteria selection**
- **Validate the selection of R_{H_Y} as the task oriented control variable and determine a meaningful aileron test input**
- **Ascertain the need for the aerodynamic force terms in the $\dot{\beta}$ estimate**
- **Demonstrate the viability of the $\hat{\beta}$ control law for the solution of the air-to-ground targeting problem.**
- **Optimize the feedback gain, denoted by $K_{\dot{\beta}}$.**

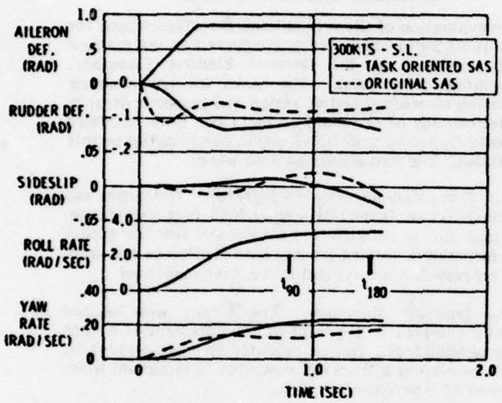

Figure 4. Maximum Roll Comparison

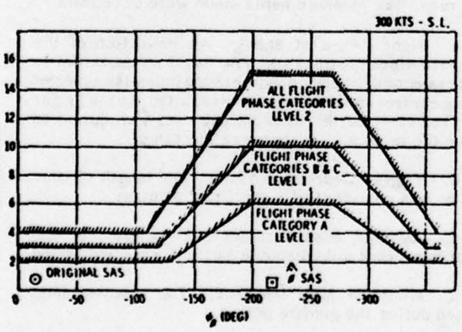

Figure 5. Sideslip Excursion Limitations

Based on analysis of the types of control inputs normally seen in the rollout and tracking phases of air-to-ground gunnery, 1- and 2-second aileron doublet input responses in R_{H_Y} were selected as criteria for SAS performance evaluation. These criteria were first applied to the original A-10 SAS for the simplified scenario previously described. The results of a half-stick doublet

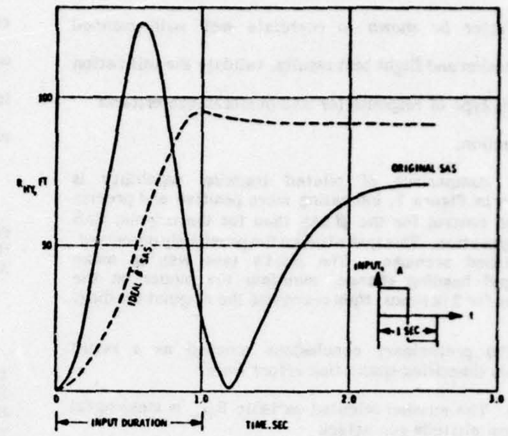

Figure 6. Response to a 1-Second, Half-Stick Aileron Doublet For Tracking Scenario

are shown in Figure 6. The time history, shows that the response of the original SAS for this particular input is both oscillatory and slow, persisting for 2.2 seconds for a 1-second duration input. The original SAS consists of a washed-out yaw rate command to the rudder for "dutch roll" damping, plus an aileron-to-rudder crossfeed to compensate for the inherent adverse yaw of the aircraft. By contrast, the response of an idealized $\dot{\beta}$SAS system is seen to be both much faster and better damped. The idealized $\dot{\beta}$ system response was obtained utilizing a feedback gain of $K_{\dot{\beta}}$ = -2 seconds which was found to optimize doublet response at the design flight condition of 300 knots at sea level. This value was used for all manned simulation and flight test evaluation. These results, which

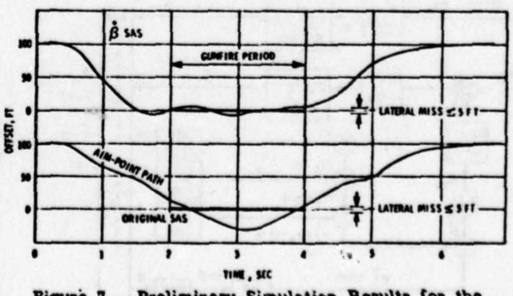

Figure 7. Preliminary Simulation Results for the Original and $\dot{\beta}$SAS Configurations

will later be shown to correlate well with manned simulation and flight test results, validate the utilization of this type of response for SAS performance criteria evaluation.

A comparison of piloted tracking capability is shown in Figure 7, indicating more positive and precise pipper control for the $\dot{\beta}$SAS than for the original SAS configuration. This task utilized the previously described, simplified scenario. The pilot's task was to make a rapid heading change, maintain the pipper on the target for 2 seconds, then reacquire the original heading.

The preliminary conclusions reached as a result of this simplified simulation effort were:

- The mission oriented variable R_{H_Y} is meaningful for low altitude gun attack
- Response to a 1-second aileron doublet is very useful for judging the relative merits of competitive SAS systems, and correlates well with piloted simulation results
- The piloted runs show that the lateral tracking error for the $\dot{\beta}$SAS is significantly less than that for the original SAS
- The results of this study were sufficiently encouraging to motivate detailed parameter selection for a manned simulation utilizing a 6-DOF model.

8. System Definition

The viability of an idealized $\dot{\beta}$SAS for providing good control of the mission oriented control variable, R_{H_Y}, was established, and, therefore, the practicalities of obtaining an acceptable estimate of $\dot{\beta}$ could be considered. The study started with consideration of the possibilities for simplification of the exact expression for $\dot{\beta}$ obtained from the side force equation. (Refer to Appendix for further details.) Consideration

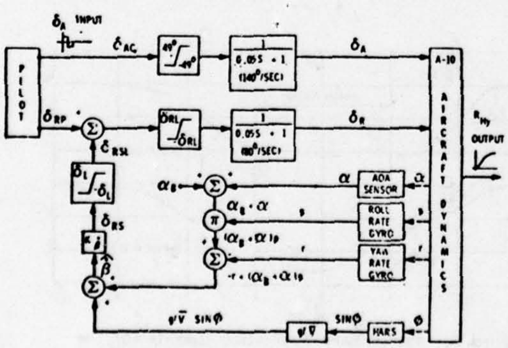

Figure 8. Task-Oriented SAS Block Diagram

of the aerodynamic characteristics of the A-10, as well as mission requirements and flight envelope, led to the selection of the following biased estimate of $\dot{\beta}$:

$$\hat{\dot{\beta}} = -r + (\alpha_\beta + \alpha)\ p + g\ \sin\phi/\overline{V}$$

Where $\overline{V}$ is an average velocity value, and α_β was chosen to be the pipper depression angle based on the reasoning presented in the system synthesis section. A block diagram for this system is presented in Figure 8.

9. Manned Simulator Studies

An evaluation of the original and $\dot{\beta}$ configurations for the A-10 SAS was subsequently conducted on the manned flight simulator at the General Electric Company, Binghamton, New York. The tasks for performance evaluation stressed speed of action and pipper control in an environment of expected ground fire. Both Fairchild Republic Company and USAF pilots participated in this evaluation. The two scenarios used were:

1. Gross Maneuver (ψ_0 = 90 degrees): The target was located initially off the left wing at 7000-feet slant range. The task was to rollin to the attack and fire the gun as soon as possible, pulling normal load factors as required. This represented the typical curvilinear maneuver

2. Tracking Maneuver: The target was located initially 400 feet to the right of gun boresight at a slant range of 5000 feet. This represented an error of 80 mils, to be removed by a tracking maneuver in minimum time, followed by a period of gunfire.

Ten runs were made for each system/scenario studied, and key measurements were taken and averaged for each set of runs. The measurements made were as follows:

a. Slant Range at Shoot: An indication of the speed with which the pilot was able to get on the target to initiate weapon delivery. A larger range indicates superior pointing control and enhances survivability. At the longer range, however, better tracking accuracy is required to achieve the same miss distance at the target

b. Trigger Depression Time: The length of time the pilot was confident of a gun firing solution

c. Average Miss Distance: A measure of the projectile average miss distance during the gunfire period

d. Minimum Miss Distance: The smallest miss obtained during the gunfire period.

Run Data and Pilot Evaluations

Average measurements for Pilots 1 and 2 are presented in Table 1, and for Pilot 3 in Table 2. It can be seen that the $\dot{\beta}$SAS is clearly superior. This conclusion is validated by the much longer ranges at gunfire initiation, combined with longer trigger times and comparable or better miss distances achieved with the $\dot{\beta}$ system. Note that Pilot 3 achieves an average miss distance of less than 10 feet in the tracking task with the $\dot{\beta}$SAS. Figures 9 and 10 are samples of the two scenarios flown by Pilot 2, and show comparisons of the two configurations. The improved lateral tracking error characteristic of the $\dot{\beta}$ system is clearly indicated, as well as the much longer range at shoot.

Table 1. Original Versus $\dot{\beta}$ SAS Comparisons

Maneuver	Tracking				ψ_0 = 90 Degrees			
Pilot	1		2		1		2	
SAS Type	Original	$\dot{\beta}$	Original	$\dot{\beta}$	Original	$\dot{\beta}$	Original	$\dot{\beta}$
Range at Gunfire (Ft)	2070	3440	2210	3810	3960	5110	2830	4230
Trigger Time (Sec)	0.8	1.3	1.3	2.9	2.0	3.1	2.0	3.2
Minimum Miss (Ft)	15	18	14	12	19	16	9	12

Table 2. Tracking Task Averages for Pilot 3

SAS Type	Original	$\dot{\beta}$
Range at Gunfire (Ft)	3090	3370
Trigger Time (Sec)	1.4	1.3
Average Miss (Ft)	24.5	9.5
Average Error (Mil)	8.9	3.1

Additionally, the simulator was flown by Pilot 3 in a variety of maneuvers at all flight speeds in order to test the $\dot{\beta}$SAS configuration. These maneuvers included wing-overs, dives, pushovers, landings, and rolling pullups, and were made in the presence of random pitch and side gusts. No undesirable flight characteristics were detected. In fact, flying appeared to be easier, and the pilot's workload less than with the original system. Throughout the test there was consistently more positive, more natural control of the aircraft with the $\dot{\beta}$ system.

Results and Recommendations

The results and recommendation of manned simulator performance evaluation are:

1. The three pilots rated the $\dot{\beta}$ system clearly superior to the original SAS

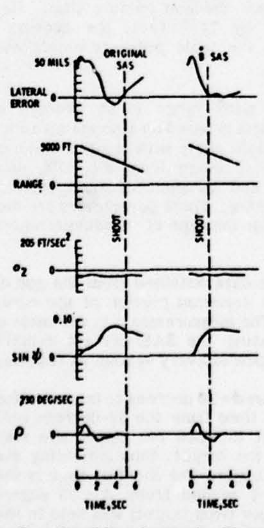

Figure 9. Tracking Scenario Comparisons

2. Superior handling qualities were reported by Pilot 3 in nonattack flying tasks for this system

3. The $\dot{\beta}$ system approximates a true, feet-on-the-floor system for attack speeds within 275-350 knots, which should provide near-optimum performance with any aircraft configuration over the attack speed range.

10. Flight Test Performance Evaluation

The real measure of performance of the SAS is the degree of proficiency with which the weapon delivery mission is accomplished. The performance of the SAS is evaluated in this section by a detailed analysis of abrupt heading changes and curvilinear strafes.

Load factor (n_z) and roll rate (p) are the most significant parameters to indicate the degree of aggressiveness with which the pilot accomplishes the curvilinear maneuver, i.e., the pilot sees the target, rolls in, and pulls up increasing the load factor to orient the aircraft toward the target as quickly as possible. During this portion of the maneuver the aircraft is banked at a large roll angle, ϕ, of between 70 and 130 degrees. Thus, the large heading change required to point at the target is accomplished primarily through pitch rate, and large n_z. As the pilot approaches the target in azimuth, the task is to roll out quickly while reducing load factor.

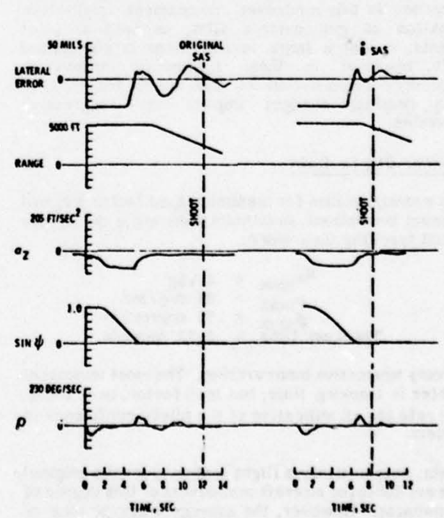

Figure 10. Gross Maneuver Scenario (ψ_c = 90 Deg) Comparisons

Table 3. Summary of Aircraft Data from Gun Camera Films

SAS Type	At ϕ = 30 Degrees				At Break			
	Airspeed (Knots)	Height AGL (Feet)	Pitch Att (Deg)	Range (Feet)	Airspeed (Knots)	Height AGL (Feet)	Pitch Att (Deg)	Range (Feet)
$\dot{\beta}$ SAS	325	1350	-13	5050	345	750	-13	3000
	330	1500	-18	4700	348	900	-17	2550
	305	1500	-19	6100	325	1050	-15	3450
	300	1400	-16	7250	322	750	-12	3200
	320	1400	-19	4450	335	800	-16	2550
Average	316	1430	-17	5510	335	850	-14.6	2950
Original SAS	305	1120	-15	3750	315	570	-13	2500

Subsequent abrupt heading change maneuvers may be required to put the pipper on target and keep it there. The ability of the aircraft to make these changes quickly, as shown by the time required prior to firing, is the true measure of SAS characteristics and performance.

Abrupt Heading Change Data

The abrupt heading change maneuver consisted of stabilizing on one target with a 10-degree dive angle, a velocity of about 300 knots, and a range of about 4000 feet, then shifting to another target as quickly as possible. Maneuver time is measured from start of rollin to shoot. The most significant parameter in evaluating system performance is the maneuver time. However, this should be normalized to account for the varying magnitude of the heading change.

Averaging data over five pilots and many maneuvers for each pilot, the normalized maneuver time is 0.82 seconds per degree of heading change for heading changes of between 4.5 and 8.5 degrees. While no quantitative flight test data is available for the original SAS performance in this maneuver environment, qualitative examination of gun camera film, as well as pilot comments, reveal a large improvement in augmented aircraft response in these aggressive maneuvers. Furthermore, examination of the times required to achieve heading changes implies very aggressive maneuvering.

Curvilinear Strafe Data

The average values for maximum load factor and roll rate during the rollout, maximum bank angle during the turn, and tracking time were:

$$n_{z_{max}} = 4.75g$$
$$p_{max} = 93 \text{ deg/sec}$$
$$\phi_{max} = 93 \text{ degrees}$$
$$\text{Tracking time} = 2.33 \text{ seconds}$$

during very aggressive maneuvering. The most important parameter is tracking time, but load factor, bank angle, and roll rate are all indicative of the pilots' confidence in the system.

Again, no quantitative flight test data for the original SAS are available for aircraft maneuvers of this degree of aggressiveness. However, the average tracking time of 2.33 seconds represents a reduction of more than 50 percent when compared with the original SAS, during significantly less aggressive maneuvers. This improvement is, perhaps, the most significant operational benefit derived from utilization of the task-oriented $\dot{\beta}$SAS.

Gun Camera Film Analysis of Curvilinear Strafes

It was decided to analyze in detail all curvilinear strafes performed by one pilot for a complete flight with the $\dot{\beta}$ SAS, and one typical maneuver by the same pilot with the original SAS.

The gun camera film material is presented in tabular and time history form. Each maneuver is picked up during rollout at the time the roll angle had diminished in magnitude to 30 degrees. The horizontal and vertical offsets represent the angular offset from the pipper to the center of the target. Positive offsets indicate the target is up and to the right.

Table 3 presents approximate airspeed, altitude, pitch angle, and slant range at two points in time for each maneuver. The first time is when the roll angle has been reduced to 30 degrees during the rollout (elapsed time of zero), and at break (when pulling off the target). The airspeed, altitude, and pitch angle readings are HUD indications read from the gun camera film. Height is altitude decreased by 2430 feet, the gunnery range altitude, therefore the table presents height as above ground level (AGL).

The estimated slant range is an average of two calculations. The first is based on aircraft altitude above ground and the gunsight angle with respect to horizontal. The second estimates range from the film, using the known target size, and its apparent size relative to the 50-mil reticle diameter. These parameters are meant to give some feeling for the type of maneuvers which were being used.

Table 4 lists the data obtained from the gun camera film records of the terminal portion of the curvilinear strafe maneuver. The measurements in the table are the criteria for evaluating the SAS/aircraft combination, comprising the weapon delivery system as follows:

1. The time from ϕ = 30 degrees to lateral offset ≤2.5 mils represents the time from the 30-degrees roll angle reference at rollout to when the pipper was stabilized within 2.5 mils of the target, thus indicating the time when gunfire could begin. The shortest time is the most desirable. Within 1 second from ϕ = 30 degrees the azimuth offset (pipper from target) was held to less than ±2.5 mils, on the average, with the $\dot{\beta}$ system. The same task could not be accomplished with the original SAS, i.e.,

Table 4. Summary of Curvilinear Strafe Performance Data from Gun Camera Films

SAS Type	Time From $\phi = 30^{\circ}$ to Lateral Offset ≤ 2.5 Mils (Sec)	Time Within ± 2.5 Mils (Sec)	Time From $\phi = 30^{\circ}$ to ϕ_{SS} (Sec)	Total Maneuver Time (Sec)	% Time Within ± 2.5 Mils	Lateral Offset at $\phi = 30^{\circ}$ (Mils)	Maximum Lateral Offset (Mils)	Time at Maximum Offset (Sec)
$\dot{\beta}$ SAS	1.78	3.6	0.50	5.4	67	0.0	5.0	0.40
	0.05	3.7	1.90	3.7	100	5.0	5.0	0.00
	0.00	3.2	0.20	3.2	100	1.0	2.5	0.40
	2.10	2.9	0.30	5.0	81	11.0	22.5	0.75
	0.58	2.6	0.50	3.6	72	25.0	25.0	0.00
Average	0.90	3.2	0.68	4.2	84	8.4	12.0	—
Original SAS	0.00	0.6	—	3.9	15	3.0	16.0	2.4–2.6, & 3.20

the pipper was offset more than 2.5 mils from the target throughout most of the maneuver

2. The time within ±2.5 mils laterally indicates the total time the pipper was held at less than 2.5 mils of the target in azimuth, and indicates vernier control in the lateral/directional mode. The longest time is the most desirable, and is 3.2 seconds for the $\dot{\beta}$SAS, and 0.6 second for the original SAS

3. The time from $\phi = 30$ degrees to ϕ_{SS} indicates the time taken to rollout, or the aggressiveness and subsequent controllability of the maneuver. The $\dot{\beta}$SAS time is 0.7 second, indicating a high level of controllability during a very aggressive maneuver. A steady-state bank angle did not occur with the original SAS, indicating that the pilot was continually maneuvering

4. The total time measures the time from roll reference (30 degrees) to pulloff from the target. This only indicates the maneuver duration

5. The percentage of time within ±2.5 mils in azimuth, which is the ratio of time within ±2.5 mils to the total time of the maneuver, measures firing opportunity. The $\dot{\beta}$SAS averaged 84 percent of the total time of the maneuver, as compared with only 15 percent for the original SAS. Note also that every maneuver with the $\dot{\beta}$ SAS was within ±2.5 mils for over two-thirds of the total time of the maneuver

6. The lateral offset at $\phi = 30$ degrees averages to 8.4 mils with the $\dot{\beta}$SAS, as compared with 3 mils with the original SAS. However, the maximum lateral offset occurring during the maneuver averages to 12 mils with the $\dot{\beta}$SAS, and is 16 mils with the original. The time to the greatest offset subsequent to the $\phi = 30$ degrees reference point is less than 1 second, indicating accurate control for the remainder of the maneuver with the $\dot{\beta}$SAS. Maximum lateral offset with the original SAS occurred from 2.4 to 2.6 seconds, and again at 3.2 seconds, indicating reduced controllability.

Time histories of typical curvilinear strafes with the original SAS and with the $\dot{\beta}$SAS are presented in Figures 11 and 12. Even though the maneuver with the original SAS is accomplished at a much lower roll rate than with the $\dot{\beta}$SAS, it is clear that the pilot has less control over pipper placement in the original configuration during this task.

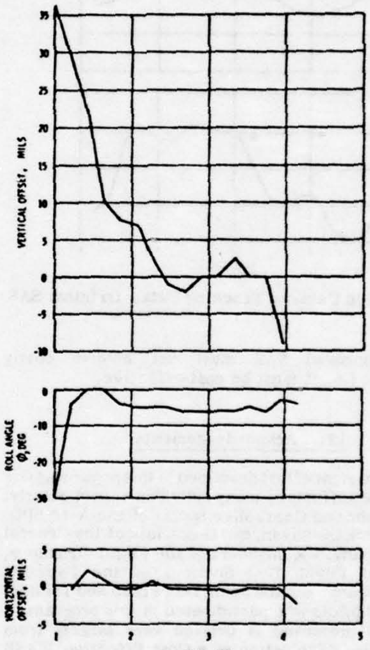

Figure 11. Gun Camera Tracking Data: $\dot{\beta}$ SAS

11. Conclusions

The concept of a task-oriented SAS seems fundamental, in retrospect, so successful was the design and implementation of the A-10 $\dot{\beta}$SAS. However, SAS design previously has been rarely motivated directly by the primary mission requirements of the aircraft, but rather by MIL-F-8785B, the generalized military flying qualities specifications.

The success of the task-oriented A-10 SAS design is indicated by the decision to incorporate the $\dot{\beta}$SAS in all subsequent production A-10 aircraft. Importantly, this success was achieved while adhering to the ground rule

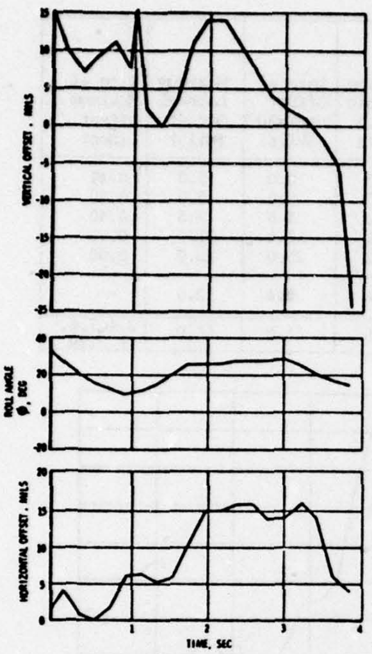

Figure 12. Gun Camera Tracking Data: Original SAS

that any improved SAS must not involve costly modifications, i.e., it must be cost-effective.

12. Acknowledgements

The development effort described in this paper was due to the combined efforts of many individuals, particularly: Mr. Wayne Thor and Capt. Mike Burski of the A-10 SPO; Ron Gague, Dick Quinlivan, and Dick Clark of the General Electric Company; the members of the Flight Dynamics, Avionics, and Flight Test Sections of the Fairchild Republic Company; and the Joint Test Force and Tactical Air Command pilots who participated in this program. The material presented is derived very largely from a paper entitled "Evaluation of a Cost Effective, TASK Oriented, Lateral Directional SAS for the A-10 aircraft" presented at the AIAA Aircraft Systems and Technology Conference in August, 1978.

13. Appendix

This appendix presents a brief outline of the various candidate estimates for $\dot{\beta}$ which were considered during the SAS design. The most complete (and costly) estimate for $\dot{\beta}$ is:

$$\hat{\dot{\beta}} = -r + (\alpha + \alpha_B)p + (g/V)\sin\phi\cos\theta + Y_A/mV$$

Cost-effectiveness was a primary goal in the design of this SAS, and thus various means of estimating $\dot{\beta}$ were considered closely so that the most cost-effective $\dot{\beta}$ estimate could be identified. All of the options considered were studied extensively via manned simulation, including both weapon delivery maneuvers and aerobatics.

The first simplification considered was the elimination of the aerodynamic side force term, Y_A. This did not affect significantly dynamic performance, and further eliminated a requirement for an accelerometer. It can also be shown by simple analysis that this result might be anticipated.

The LaPlace transformation of the side force equation can be written as:

$$\dot{\beta}(s) = s/(s + \bar{Y}_\beta)\,\dot{\beta}_0(s)$$

Where: $\bar{Y}_\beta = 0.20$ (at 300 knots, sea level)

$$\dot{\beta}_0(s) = -r(s) + (\alpha + \alpha_B)\,p(s) + (g/V)\,\phi(s)$$

It can be seen, on a linear basis, that the effect of dropping the side force term is to replace the term:

$$F(s) = s/(s + 0.20) = 5s/(5s + 1)$$

by unity. However, F(s) is seen to be a long time constant washout compared with the duration of tactical maneuver elements and will not significantly affect dynamic response during these maneuvers. Then, the estimate for $\dot{\beta}$ becomes:

$$\hat{\dot{\beta}} = -r + (\alpha + \alpha_B)\,p + (g/V)\,\sin\phi\cos\theta$$

Next, it was recognized that, in a low-altitude, tactical maneuver, the pitch angle is rarely large, so that the approximation $\cos\theta = 1$ should be evaluated. This approximation will produce a rudder command in error by $\epsilon_{\delta R}$, where:

$$\epsilon_{\delta R} = K_{\dot{\beta}}g\,\sin\phi\,(\cos\theta - 1)/V$$

It can be seen that this error will not be significant unless both roll and pitch angles are simultaneously large. This approximation does not have any noticeable effect on the tactical maneuvering simulation results, and did not produce unacceptable dynamic response, even for rolling maneuvers at approximately 90-degree pitch attitude. Based on these results, the estimate for $\dot{\beta}$ was taken to be:

$$\hat{\dot{\beta}} = -r + (\alpha + \alpha_B)\,p + g\,\sin\phi/V$$

The replacement of the g/V term with the piecewise constant $\bar{V}$ is described in the text. This approximation leads to the final form for the $\dot{\beta}$ estimate of:

$$\hat{\dot{\beta}} = -r + (\alpha + \alpha_B)\,p + g\,\sin\phi/\bar{V}$$

with which excellent flight test results were obtained. Other, still simpler, estimate schemes involving the replacement of the term $(\alpha + \alpha_B)$ p with a term $\bar{\alpha}$p, where $\bar{\alpha}$ is a constant, were also considered, but pilot preference led to the selection of the estimate presented above. This estimate is the simplest and least costly estimate which led to consistently good dynamic performance and pilot acceptance, and therefore was deemed to be the most cost-effective.

QUESTIONS AND ANSWERS

1. Dwight Schaefer - Boeing:

Question: β includes an a_y term. Was the effect of this term examined? What were the Dutch Roll stability characteristics for the two systems? Were the problems associated with the first yaw damper due to turn coordination, not Dutch Roll.

Answer: The effect of the a_y term was studied in detail. Both analytical and simulator studies were conducted. The conclusion was that, for the rapid maneuvers being considered, a_y feedback did not make any significant difference in dynamic response, i.e., β feedback was not required. The problems associated with the first yaw damper were caused by the nature of the sideslip response caused by the ARI and not due to Dutch Roll damping.

2. Chick Chalk, Calspan:

Question: You changed the phase of the sideslip, do pilots use the rudder?

Answer: Yes they generally still use the rudder but to a much smaller extent than they previously did. In fact, the dominant pilot comment in this area was that they had to relearn rudder technique and use much less rudder and furthermore to use it in a pulsed mode, i.e., get on and off the pedals very quickly.

3. Wayne Thor, ASD:

Question: Did the pilots rate the original SAS as level 1?

Answer: The lateral-directional SAS met 8785B level 1 for attack flight conditions. It did not meet level 1 for this scenario.

4. Bill Lamar, AFWAL:

Question: Have you considered additional degrees of freedom?

Answer: Yes, control of additional degrees of freedom has been and is being considered in order to enhance air-to-ground attack. Preliminary studies have shown that direct force can be very useful for enhancing this attack mode.

5. Tom Twisdale, AFFTC:

Question: Why didn't the problem with rapidly acquiring the target after a flight path correction surface in early simulation studies? Why did you have to wait till flight test to see it?

Answer: Early simulation studies focused on the originally planned tactical maneuvers which were satisfactorily accomplished. It was only as the maneuvers became very aggressive that the problem surfaced.

LONGITUDINAL MANEUVERING CONTROL CHARACTERISTICS OF A CANARD-WING FIGHTER CONFIGURATION

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Analyses are made of a canard-wing fighter configuration, each surface having a trailing edge flap, to determine the maneuvering and tracking capability. Variations in the relationship between the canard and wing flap deflection allow longitudinal control concepts to range from a pure pitch control (pure pitching moment change) to a pure direct lift control (pure lift change at constant angle of attack). Results from a piloted tracking simulation are given and compare direct lift control for maneuvering and tracking to pitch control concepts and to an aft tail configurations. A reduction in tracking error is shown for the direct lift control mode. Evaluations of the direct lift control mode are made with reference to MIL-F-8785B, Military Specification - Flying Qualities of Piloted Airplanes. The specification problems encountered in assessing the direct lift control mode are discussed.

NOMENCLATURE

$C_{L_{\delta_c}}$	Canard lift with δ_c
$C_{L_{\delta_w}}$	Wing lift with δ_w
$C_{L_{\delta}}'$	Total control lift
$C_{M_{\delta_c}}$	Canard pitching moment with δ_c
$C_{M_{\delta_w}}$	Wing pitching moment with δ_w
$C_{M_{\delta}}'$	Total control moment
C_{L_α}	Airplane lift curve slope
$(C_L)_{1g}$	$W/\bar{q}s$
g	Acceleration of gravity
m	Airplane mass
M_δ'	Total control pitching moment derivative
M_w	Airplane pitching moment derivative due to w
M_α	$M_w V$
M_q	Pitch damping derivative due to q
$M_{\dot{w}}$	Pitch damping derivative due to $\dot{w}$
$M_{\dot{\alpha}}$	$M_{\dot{w}} V$
N_z	Normal acceleration in g units
q	Pitch rate
$\bar{q}$	Dynamic pressure
s	Reference area (wing)
V	Total velocity
w	Vertical velocity
Z_δ'	Total control normal force derivative
Z_w	Normal force derivative due to w
α	Angle of attack
δ_c	Canard flap deflection
δ_w	Wing flap deflection
δ'	Reference deflection (same as δ_c)
$\omega_{N_{SP}}$	Short period frequency
ζ_{SP}	Short period damping factor
$\ddot{\theta}$	Pitch acceleration
τ_{θ_2}	Lead factor in θ/δ transfer function

Special Subscripts

$t=0^+$	Evaluated with initial value theorem
ss	Evaluated with final value theorem

INTRODUCTION

The altering of aircraft flight paths by direct force as opposed to angle of attack change through pitching has come into increased interest. It has been investigated in the past in some studies in the form of jet thrust and wing flap deflection for increased combat normal and transverse g capability and also for flight path control in the approach to landing (reference 1 to 4). Reference 5 and 6 analytically examined direct lift control for a wide range of applications. Past investigations of direct lift control applications have generally been for wing-tail configurations.

The interesting aspect of direct lift control, as produced by a flapped surfaces, is the immediate acceleration force available limited only by the rate of flap deflection and the aerodynamic circulation lift build-up. Still another interesting aspect is the capability to produce flight path changes without the need to rotate the entire aircraft to increase angle of attack to obtain the accelerative lift. Some studies such as reference 7 have used a form of direct lift control to point an aircraft while essentially at constant lift.

A canard-wing aircraft configuration lends itself well to a study of aircraft pitch control and direct lift control. With effective trailing edge flaps on both the canard surface and the wing surface, a range of control relationships from a powerful pitching moment control to a direct lift control can be investigated. In this study a pitching moment control system is defined as one in which deflection of the canard and wing flap deflections are opposite to produce summing moments and opposing lifts. A direct lift control is defined as one in which the canard and wing surfaces deflect in the same direction to produce opposing moments but summing lifts.

It is possible to produce a pure pitching moment control in which the net lift due to the canard and wing flap deflections is zero. Conversely, a pure direct lift control is one in which deflection of the canard and wing flaps produce zero pitching moment change. A perfect direct lift control is defined as one in which the steady state angle of attack change with normal acceleration is zero.

The specification MIL-F-8785B, Flying Qualities of Piloted Airplanes, however, has developed over the years based on experience, analysis and studies of conventionally controlled (generally wing-tail) airplanes in pitch, i.e., flight path changes accomplished with angle of attack change through rotation in pitch. While the current specification acknowledges the concept of direct lift control, quantification and definition of assessment parameters is clearly lacking. Therefore, by use of the variability offered by a canard-wing configuration, various degrees of direct lift control are examined as to capability and the relationship

to MIL-F-8785B, Flying Qualities of Piloted Airplanes. The study is exploratory to examine what can be accomplished with direct lift control and what kinds of problems might be encountered.

CANARD-WING CONFIGURATION DESCRIPTION

The configuration examined is shown in Figure 1 along with the important geometric constraints. This configuration offers some unique aerodynamic features. One feature is the excellent pitch damping resulting from the fact that both the wing and canard centers of lift due to angle of attack are located a good distance from the aircraft center of gravity. As a result the basic airframe short period damping is excellent as will be shown later.

The pitch and/or direct-lift control is obtained from the canard surface trailing edge flap and the wing surface trailing edge flap which in the latter also serves as the roll control. Either the canard or wing flap can produce large pitching moments due to the relatively large distance from the c.g. of their centers of pressure due to deflection.

The wing deflection is linearly related to the canard deflection by

$$\delta_W = \frac{\delta_W}{\delta_C}\,\delta_C + \delta_{WO} \tag{1}$$

The effective values of the total airplane lift and moment pitch control effectiveness are obtained as follows:

$$C_{L_\delta}' = C_{L_{\delta_C}} + C_{L_{\delta_W}}\,\frac{\delta_W}{\delta_C} \tag{2}$$

$$C_{M_\delta}' = C_{M_{\delta_C}} + C_{M_{\delta_W}}\,\frac{\delta_W}{\delta_C} \tag{3}$$

Variation of the linear gearing, (δ_W/δ_C), and both positive and negative values allows a large range of effective values of C_{L_δ}' and C_{M_δ}' to be achieved. The deflection δ' is referenced to the canard flap deflection in equations (2) and (3), above. Figure 2 illustrates the above control effectiveness values for the three specific Mach numbers considered. The value of δ_{WO} (equation 1) is used to bias the wing deflection to place the canard and wing surface deflection range at reasonable values with respect to the maximum and minimum deflections.

Maneuvering Characteristics

The typical maneuvering characteristics are illustrated for a transonic combat flight range at 20,000 ft for a gross weight of 18,500 lbs at a

nominal center-of-gravity. The basic airframe dynamic characteristics are presented in Figure 3 and are, of course, independent of the control relationships. Figure 4 illustrates the variations in M_δ' and Z_δ' as a function of canard-wing gearing.

The steady state normal acceleration values available as a function of the canard-wing gearing are shown in Figure 5. The positive values of the obscissa in Figure 4 represent direct lift control gearings while the negative values are pitch control values. As can be seen in Figure 5, the normal acceleration per unit deflection decreases as the canard-wing gearings tend toward direct lift controls. The incremental values of the canard deflection to maneuver to the assumed maximum normal acceleration are shown in Figure 6. It is evident in Figure 6 that large deflections are required for high levels of direct lift control which can limit the maneuvering envelope. This can be alleviated to some degree by biasing the wing deflection at neutral control to shift the range of canard deflections but possibly at the expense of the opposite end of the envelope at some flight conditions and center-of-gravities.

The reciprocal of the normal acceleration sensitivity is presented in Figure 7. Very little change can be seen in Figure 7 for the normal acceleration sensitivity for a relatively wide range of wing to canard deflection ratios. For values of $\alpha/N_z = 0$, the quickening of normal acceleration response due to control input should be at its maximum since no steady state angle of attack change is necessary. It is noted that at the low Mach number of .6, achieving low values of α/N_z with δ_W/δ_C gearings of about .6 is not feasible since, as Figure 5 indicates, little normal acceleration can be produced.

<u>Tracking Simulation Description</u>

The most useful application for direct lift control would appear to be in target tracking. It is in this flight task that the most demanding requirements for pitch control response exists. A five degree-of-freedom piloted simulation was conducted to determine the effect of the type of longitudinal control system on the air-to-air tracking task. The pitch control configuration consisted of an aft tail pitch control and various combinations of canard-wing pitching moment control and direct lift control as discussed earlier. The simulation evaluations were performed at Mach = .6, .9 and 1.3. The altitude range was 10,000 feet to 35,000 and each run was initiated at 20,000 feet.

The simulation was conducted on the Rockwell International Corp., Columbus Aircraft Division's Dynamic Flight Simulator which utilizes a moving base cockpit and a projected television visual display for the out-of-cockpit view. A fixed reticle sight mounted at the wind screen was used to sight and track the target aircraft. The target was a scale aircraft model mounted on an angular table system which was driven by

the computer. The target motions were obtained by fighter pilots flying the cockpit of the Dynamic Flight Simulator in mock evasive maneuvers using the cockpit instrument panel. The target Euler angles and spacial coordinates were recorded during the evasive maneuvering flights for subsequent playback during the evaluation phase.

The five degrees-of-freedom aircraft equations of motion were programmed on hybrid computing equipment with Mach number held constant during a run. The taped Euler angles and spacial coordinates of the target aircraft were played back and similar real time computed data for the pursuing aircraft were used to present a pictorial image on the visual display to the pilot. The pilot then tracked the target by flying the cockpit to keep the gun sight pipper on the target. The range to the target was held constant. Since relatively long term accelerations were involved, cockpit motions could not be utilized. However, cockpit buffet was simulated to warn of approaching stall was produced as a function of normal acceleration and altitude at each Mach number. Pilot cues other than buffet were provided by the visual presentation of the target airplane, the cockpit instruments and the cockpit control forces.

Good lateral-directional aircraft characteristics were employed with appropriate stability augmentation so that the emphasis could be put on the longitudinal control. Excellent roll response was available at all speeds. No longitudinal stability augmentation was employed. The basic airframe longitudinal short period frequency and damping which were considered satisfactory for the evaluation is shown in Figure 3 for the mid-altitude of 20,000 ft.

The longitudinal control configurations evaluated consisted of four canard-wing relationships. In addition, a wing tail configuration was evaluated. Table I shows the wing to c nard ratios and the wing-tail configuration elevator characteristics evaluated.

Table I.

δ_w/δ_c	M	N/α g/RAD	N/δ g/RAD	Configuration
-.35	.6	17.37	92.27	Canard-Wing
	.9	44.73	77.49	"
	1.3	82.07	75.35	"
.2	.6	17.4	39.24	"
	.9	45.5	38.96	"
	1.3	84.0	36.95	"
.48	.6	25.02	12.39	"
	.9	70.03	19.48	"
	1.3	171.67	17.42	"
.6	.9	163.3	11.09	"
	1.3	-546.2	9.05	"
-	.6	17.37	18.16	Wing-Tail
-	.9	44.73	69.84	"
-	1.3	82.07	69.73	"

The wing-tail configuration was evaluated using the same wing as the canard-wing configuration.

The stick force per g values were maintained within a range of 4.5 to 6 lbs/g for 20,000 ft as shown in Figure 8. The range of stick force per g values were considered optimum for the tracking evaluations. Stick deflection sensitivity was maintained at reasonable values for all control configurations.

Tracking Simulation Results

The simulator tracking evaluations were flown by 5 combat rated fighter pilots. The various configurations were flown by them in a random order. Each pilot also rated each control configuration.

The step response for each control configuration is shown in Figure 9 and were obtained from the simulation. For .6 Mach number in Figure 9, note that the highest wing-to-canard deflection ratio evaluated was δ_W/δ_C = .48 due to the high deflections required for values of δ_W/δ_C greater than .48 at that flight condition. Examination of the N_Z and α traces shows that for δ_W/δ_C = .48 only a small reduction in angle of attack and small improvement in normal acceleration response can be noticed. Significant improvement for the values of δ_W/δ_C = .48 and .6 can be seen for the .9 and 1.3 Mach number responses in Figure 9. The normal initial N_Z reversal for the wing-tail configurations at all Mach numbers can be seen in Figure 9. This, of course, is due to the downward direct normal force from the horizontal tail which is used to pitch the aircraft up. It does, however, result in a delay in N_Z increase for approximately .1 to .2 seconds for the step responses of Figure 9. No delay occurs for the canard-wing cases with δ_W/δ_C = -.35 and -.2 and in immediate increase in N_Z is apparent for δ_W/δ_C = .48 and .6 as illustrated in Figure 9.

The target sighting errors which were averaged for all pilots for each control configuration are shown in Figure 10. No improvement in sighting error is apparent for the .6 Mach number case as δ_W/δ_C values become negative. The .9 and 1.3 Mach number show reductions in sighting error as direct lift control values of δ_W/δ_C are approached in Figure 10. It is significant that the greatest reduction in sighting error was obtained at the highest Mach number evaluated.

A typical distribution in tracking error is shown for 1.3 Mach number in Figure 11 and clearly shows the improvement in error as the control tends to greater degrees of direct lift control.

The tracking error was also separated into lateral and vertical errors as the evaluation were flown. Generally, the lateral errors were about the same order of magnitude as the vertical errors. However, for the control configuration values of δ_W/δ_C = .48 and .6 for 1.3 Mach number, the vertical errors were about 15% less than the lateral errors.

The averaged pilot ratings were presented in Figure 12 and show approximately the same trends as the sighting error performance. The low range of the pilot ratings is due to several overall factors in the simulation the pilots disliked. These factors being the lack of a horizon reference in the outside visual view, the narrow field of view and the lack of additional "G" cues other than stick force and buffet near stall. However, the important feature to be shown with the pilot ratings is that as the mode of longitudinal control changed from pitch control to direct lift control and its rapid response, no degradation in pilot ratings occurred. Even at the Mach number of 1.3 for the direct lift control with the steady state $\alpha/N_Z = 0$ the pilot ratings were relatively good.

It was concluded, like previous analytical studies conducted at the Columbus Aircraft Division of Rockwell International Corporation had shown, direct lift control could improve longitudinal maneuvering characteristics. It is not clear yet just how direct lift control should be implemented. In view of the large flap deflections and possible drag penalties incurred, the best form of application must be further studied. Perhaps direct lift control used for short term precision maneuvering or a form that utilizes the initial response characteristic which then washes out might be the best use. In any event, forms of direct lift control will most likely find usage in future military aircraft.

MIL-F-8785B CONSIDERATIONS

As is known the present requirements in regard to longitudinal control in the specification MIL-F-8785B are based on the wealth of data and analyses available for wing-tail configurations. Use of MIL-F-8785B as guidance for design of direct lift control systems was, of course, not intended and the current specification acknowledges this. Use of the parameter N_Z/α is the fundamental problem. One such use of the parameter N_Z/α is in the MIL-F-8785B requirement for ω_{NSP}. Surprisingly, many of the direct lift control configurations evaluated in the simulation fall in the level one region as seen in Figure 13. However, the case of 1.3 Mach number with $\delta_W/\delta_C = .6$ cannot be plotted in Figure 13 since N_Z/α is a large negative number (small α/N_Z).

The maneuvering stick force gradient specification requirement is also in terms of N_Z/α in MIL-F-8785B. It presents less of a problem applied to direct lift control systems since at high values of N_Z/α the stick force maximum and minimum gradients are specified as a constant. Figure 14 shows the force gradients used in the simulation compared to MIL-F-8785B boundaries. Values cannot be shown for 1.3 Mach number since the value of N_Z/α is a large negative number as for the ω_{NSP} comparison.

The analytical expression for N_Z/α can be examined as follows:

$$\frac{N_Z}{\alpha} = \frac{V}{g}\left[\frac{Z_\delta{}'\,M_w - M_\delta{}'\,Z_w}{M_\delta{}' - \dfrac{Z_\delta{}'\,M_q}{V}}\right] \qquad (4)$$

For conventional wing-tail configurations or whenever the prodominate terms in equation (4) are $M_\delta' Z_w$ (numerator) and $Z_\delta' M_q/V$ (denominator) then equation (4) reduces to

$$\frac{N_z}{\alpha} = \frac{V}{g}\frac{(-M_\delta' Z_w)}{M_\delta} = \frac{V}{g} Z_w = \frac{C_{L_\alpha}\,\bar{q}s}{m\,g} = \frac{C_{L_\alpha}}{(C_L)_{1g}} \qquad (5)$$

which is a function of the lift curve slope and flight condition embodied in $(C_L)_{1g}$. For the pure direct lift control case ($M_\delta' = 0$) the normal acceleration sensitivity is then given by

$$\frac{N_z}{\alpha} = \frac{-V^2}{g}\frac{M_w}{M_q} = \frac{-V}{g}\frac{M_\alpha}{M_q} \qquad (6)$$

which is a function of the aircraft longitudinal stability and pitch damping. It is noted that the case of $M_\delta' = 0$ is not of primary interest.

If a perfect direct lift control system is assumed, i.e., $\alpha/N_z = 0$, (or $N_z/\alpha = \infty$) then

$$\frac{\alpha}{N_z} = 0 = M_\delta' - \frac{Z'_\delta M_q}{V} \qquad (7)$$

or

$$M_\delta' = \frac{Z_\delta' M_q}{V} \qquad (8)$$

In the case of the canard-wing configuration, δ_w/δ_c can be chosen to satisfy equation (8). In this case, as expected, N_z/α is no longer a function of C_{L_α}.

Intuitively, it would be expected that for $\alpha/N_z = 0$, N_z/δ would be a function only of Z'_δ. This can be shown as follows. For the steady state

$$\frac{N_z}{\delta} = \frac{V}{g}\frac{(Z_\delta' M_w - M_\delta' Z_w)}{\omega_{N_{SP}}^2} \qquad (9)$$

Replacing M_δ' with equation (8)

$$\frac{N_z}{\delta} = \frac{-Z_\delta'}{g}\frac{(Z_w M_q - M_m V)}{\omega_{N_{SP}}^2} \qquad (10)$$

and since

$$\omega^2_{N_{SP}} = Z_w M_q - M_w V \tag{11}$$

then

$$\frac{N_z}{\delta} = \frac{-Z_\delta'}{g} \tag{12}$$

It is also of interest to examine the development of the current MIL-F-8785B boundary $\omega^2_{N_{SP}}/N/\alpha$. One approach was the ratio of the initial pitch acceleration response to the steady state normal acceleration response. The expression for the above ratio, from reference 8, is written for constant speed equations of motion by applying the initial value theorem to the $\ddot{\theta}/\delta$ transfer function and the final value theorem to the N/δ transfer function.

$$\frac{\left.\frac{\ddot{\theta}}{\delta}\right|_{t=0^+}}{\left.\frac{N}{\delta}\right|_{ss}} = \frac{\omega^2_{N_{SP}}}{\frac{V}{g}\,\frac{1}{\tau_{\theta_2}}} \tag{13}$$

where the lead factor

$$\frac{1}{\tau_{\theta_2}} = \frac{Z_\delta M_w - M_\delta Z_w}{M_\delta + Z_\delta M_{\dot{w}}} \tag{14}$$

Applying the expression for a perfect direct lift control (equation 8) the lead factor reduces to

$$\frac{1}{\tau_{\theta_2}} = \frac{\omega^2_{N_{SP}}}{-M_q - M_{\dot{\alpha}}} \tag{15}$$

Thus, for a perfect direct lift control the response ratio is

$$\frac{\left.\frac{\ddot{\theta}}{\delta}\right|_{t=0^+}}{\left.\frac{N}{\delta}\right|_{ss}} = \frac{(-M_q - M_{\dot{\alpha}})}{V/g} \tag{16}$$

Reference 8 shows that for conventional wing-tail airplanes with pitch control systems

$$\frac{\left.\frac{\theta}{\delta}\right|_{t=0^+}}{\left.\frac{N}{\delta}\right|_{ss}} \approx \frac{\omega_{N_{SP}}^2}{N/\alpha} \quad (17)$$

and it is the ratio $\omega_{N_{SP}}^2/(N/\alpha)$ which forms the boundaries currently in MIL-F-8785B for the $\omega_{N_{SP}}$ requirement. If equation (17) is erroneously used for a perfect direct lift control, it would yield a value of zero. However, as equation (16) shows, a value does exist as it intuitively should since there is a small pitch transient even for a perfect direct lift control. This implies that the lower boundaries for the $\omega_{N_{SP}}$ vs N/α requirement in MIL-F-8785B cannot be met especially for direct lift control systems where α/N_z approaches 0. Indeed, the current MIL-F-8785B specification allows the boundary on $\omega_{N_{SP}}$ and N/α to be relaxed, with the approval of the procuring activity, if a suitable type of direct-lift control is provided.

In essence, the above discussion has aimed at illustrating that the background of pitch control systems understandably so prevalent in the MIL-F-8785B specification of longitudinal maneuvering requirements is inappropriate for direct lift control systems. The current requirements may be satisfactory in some cases for direct lift control systems that are a combination of pitch control and direct lift. However, for direct lift control approaching values of $\alpha/N = 0$, new parameters need to be developed to adequately specify desirable flying qualities. Perhaps maneuvering control force gradients would be better specified in terms of N/δ for direct lift control, being somewhat analagous to N/α. Continued interest and flight testing of direct lift control applications should provide the needed data for further development.

Conclusions

A canard-wing aircraft configuration with a canard flap and a wing flap provides a wide range of longitudinal maneuvering control concepts. The relationship between the canard flap and wing flap deflections with cockpit control deflection was evaluated in a fighter target tracking simulation. Target tracking error showed continuous reduction at .9 and 1.3 Mach number as greater degrees of direct lift control were provided. At .6 Mach number sufficient levels of direct lift control could not be achieved due to the large flap deflections required at the altitude evaluated. Pilot ratings were as good for direct lift control that utilized little to no angle of attack (steady state) change even though the $\omega_{N_{SP}}$ vs N/α specification boundary was not satisfied.

The present MIL-F-8785B specification parameter, N/α, used for longitudinal maneuvering control, is not applicable to direct lift control concepts that approach a perfect direct lift control ($\alpha/N = 0$). Control force gradients might be better specified in terms of N/δ instead of N/α for perfect direct lift control concepts.

The form of the application of direct lift control for maneuvering is not clear yet. It may be better utilized for small normal acceleration changes or in a control system that washes out the direct lift control used in conjunction with a pitch control.

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Wing Area = 293 ft^2

Reference Chord (Wing = 12.5 ft

Wing Span = 24.8 ft

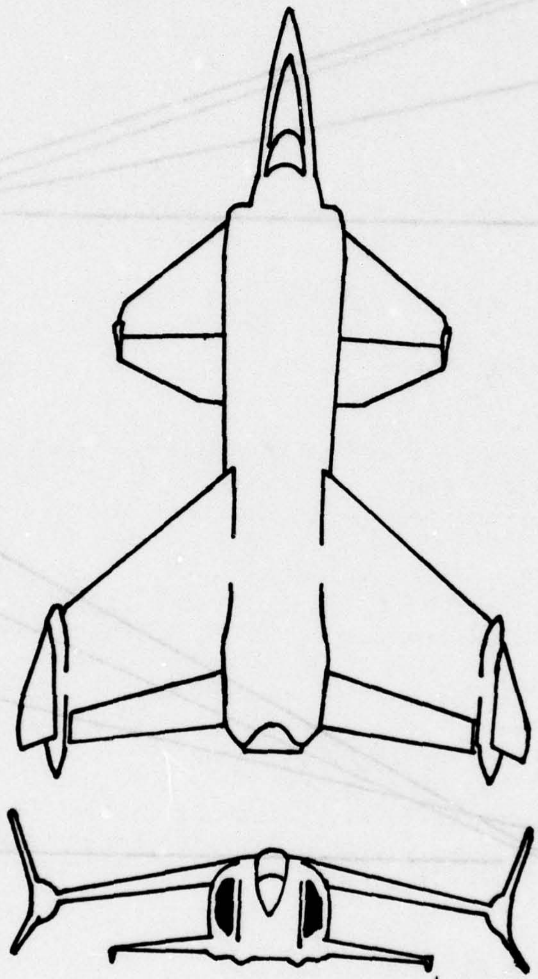

Figure 1. Canard-Wing Configuration

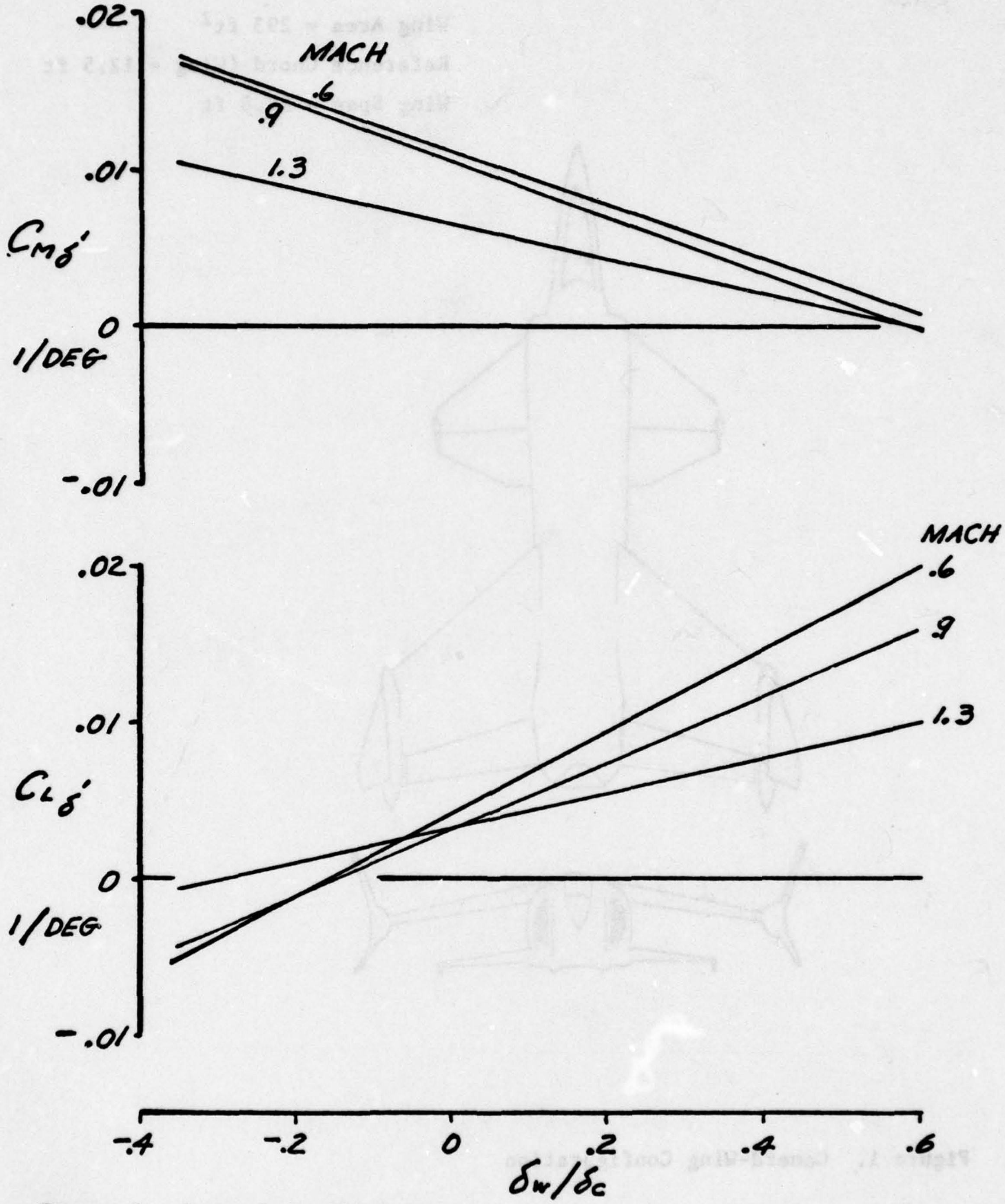

Figure 2. Control Coefficients

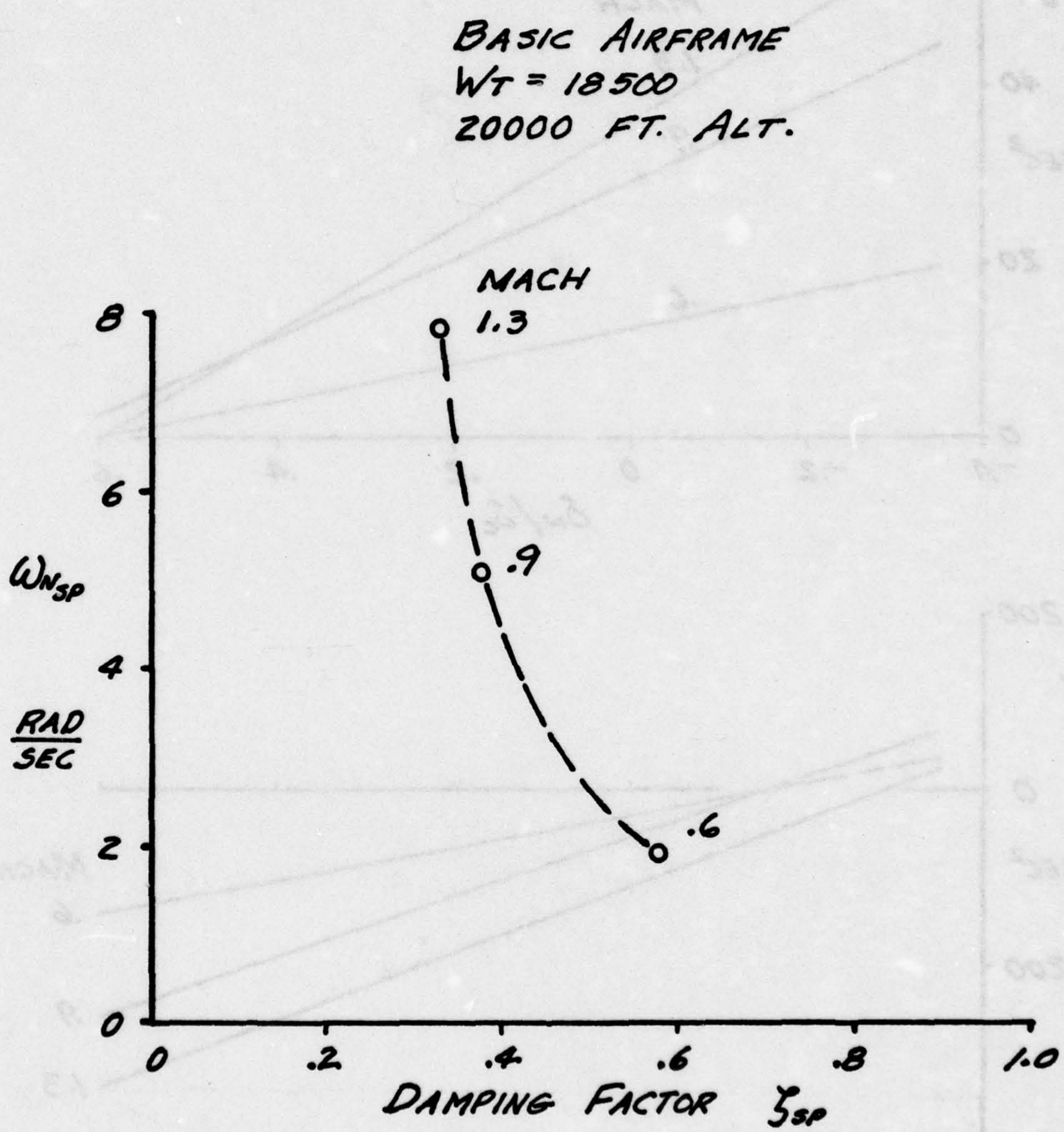

Figure 3. Longitudinal Short Period Frequency and Damping

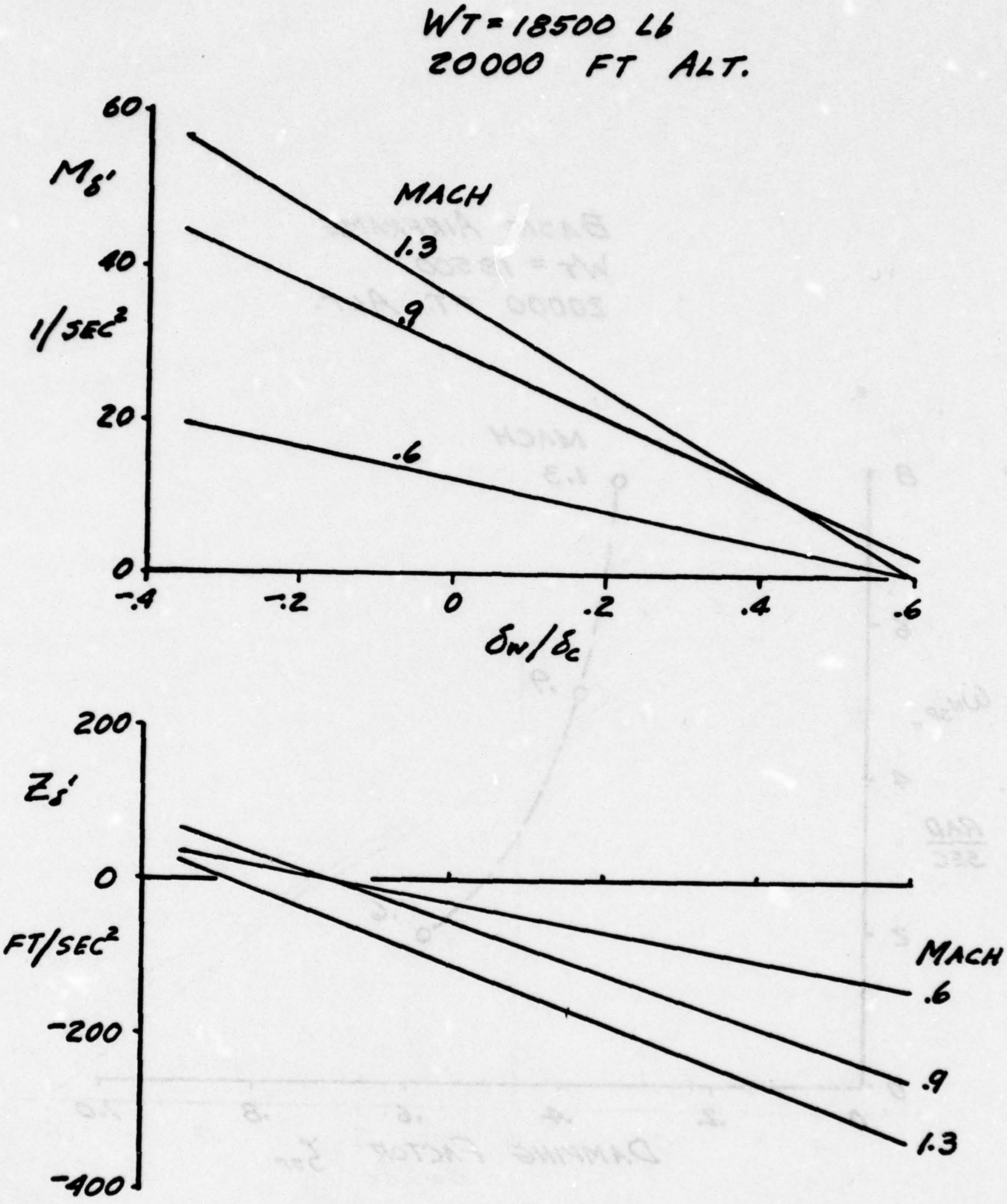

Figure 4. Control Dimensional Derivatives

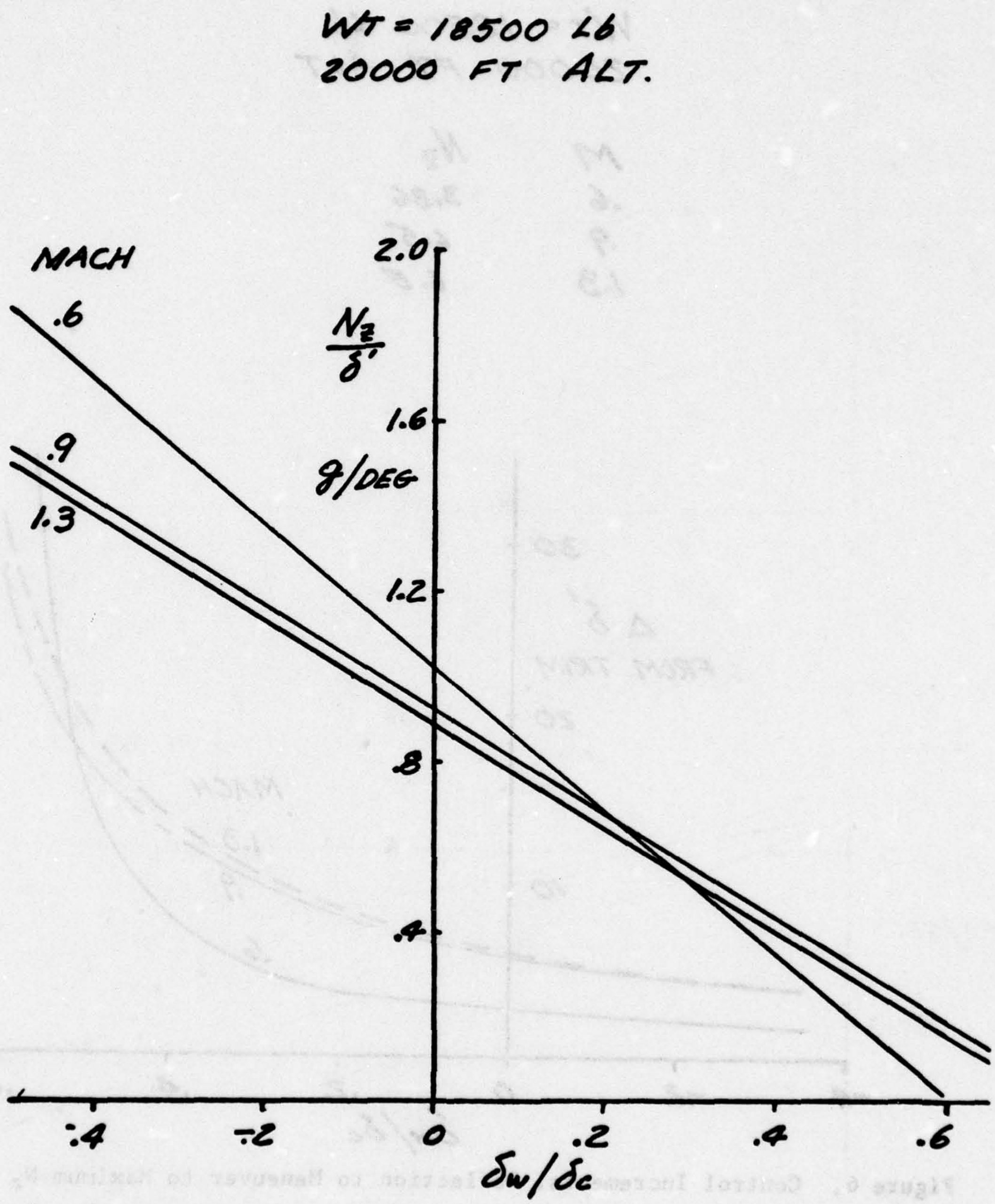

Figure 5. Normal Acceleration due to Control Deflection

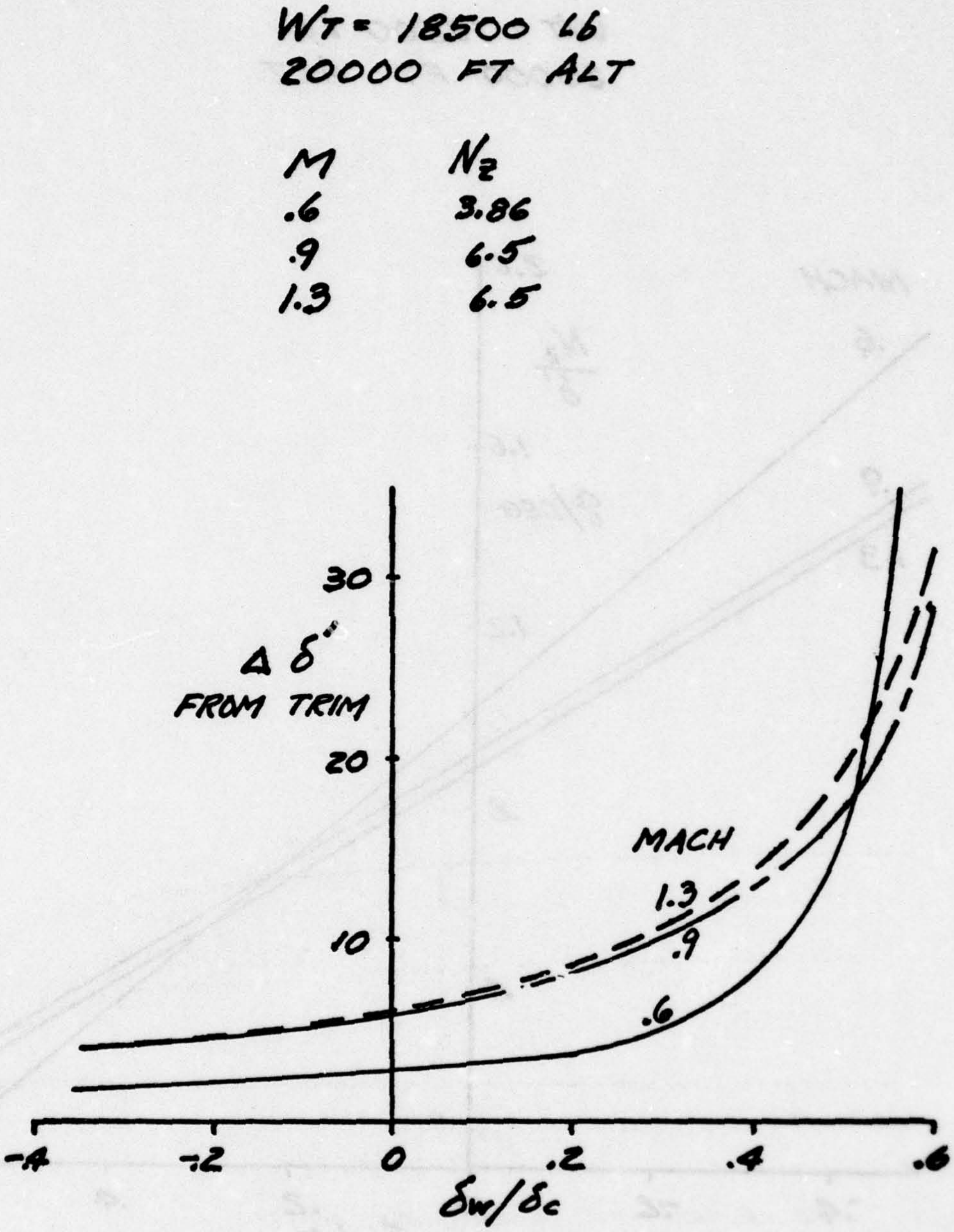

Figure 6. Control Incremental Deflection to Maneuver to Maximum N_z

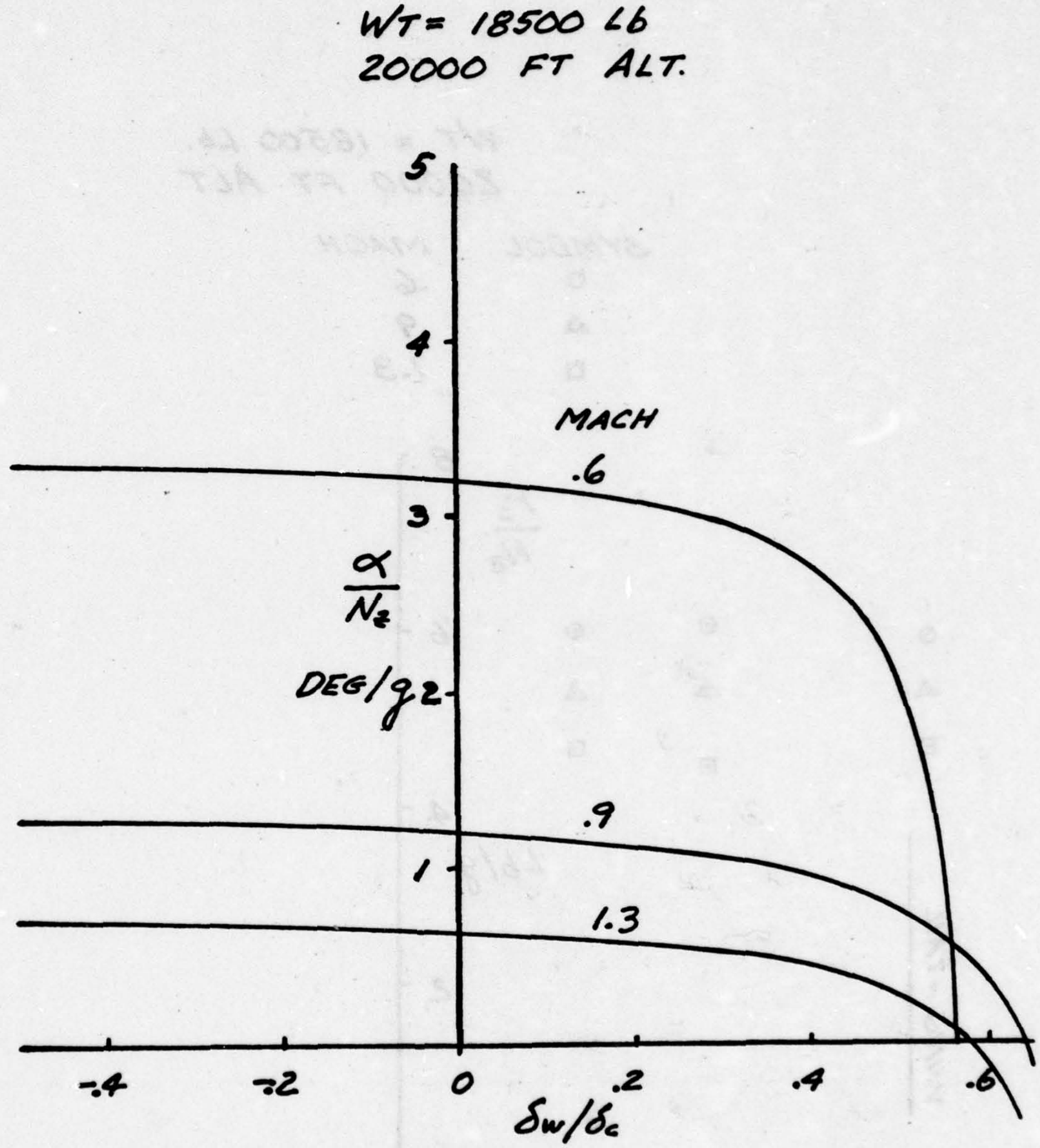

Figure 7. Normal Acceleration Sensitivity Reciprocal

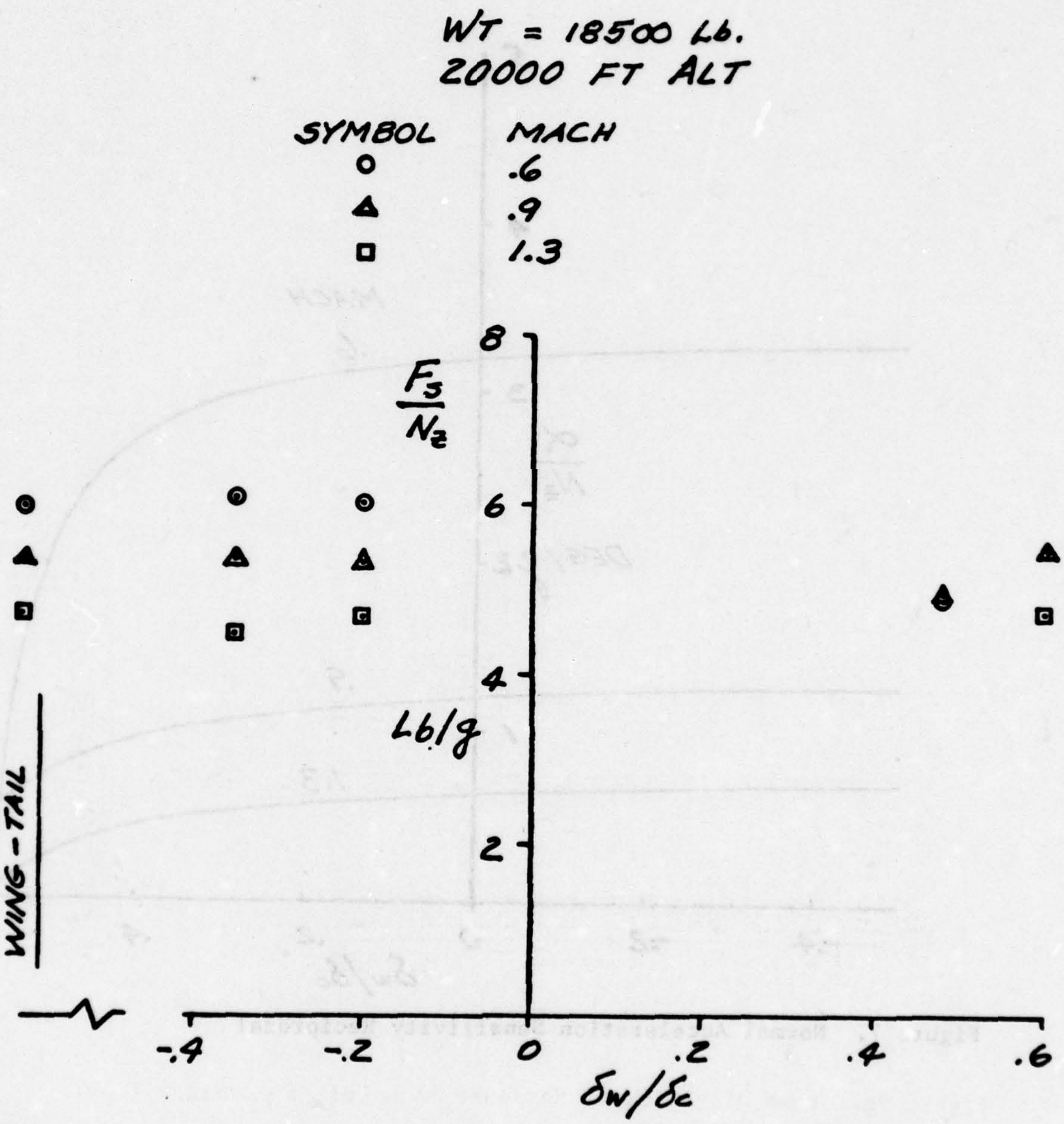

Figure 8. Maneuvering Force Gradients for Tracking Simulation

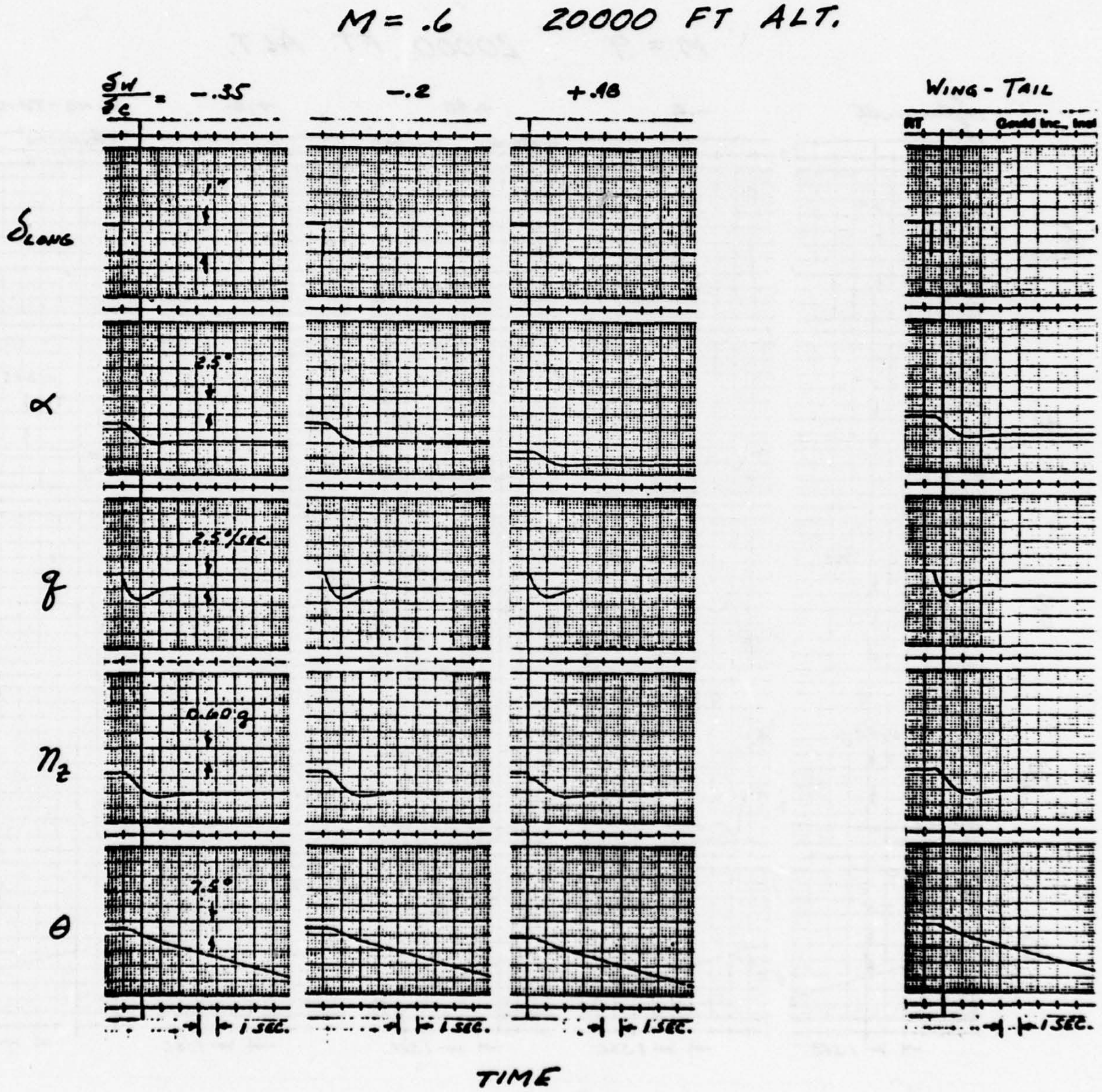

Figure 9a. Time Histories of Response to a Unit Step Stick Input

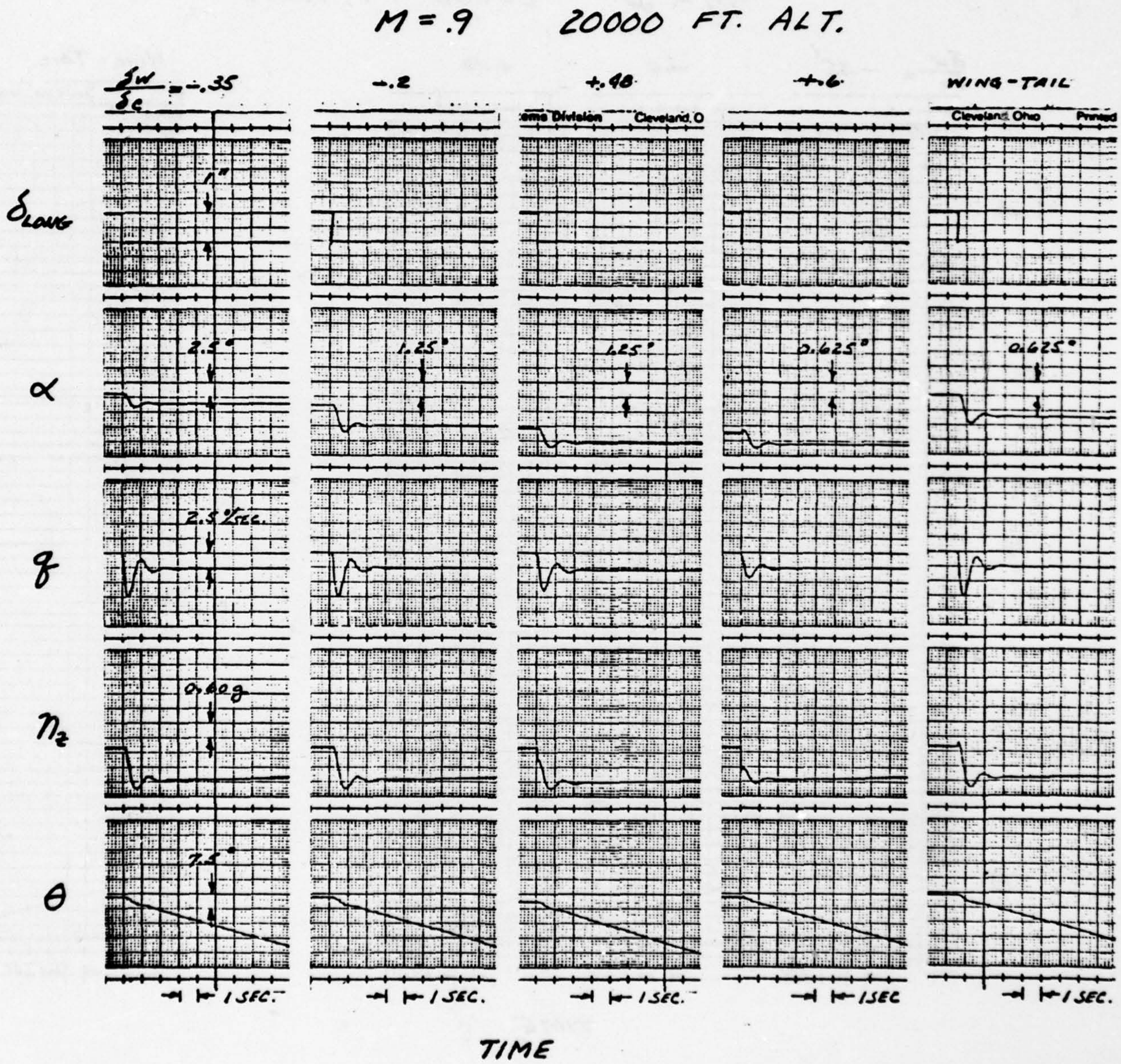

Figure 9b. Time Histories of Response to a Unit Step Stick Input

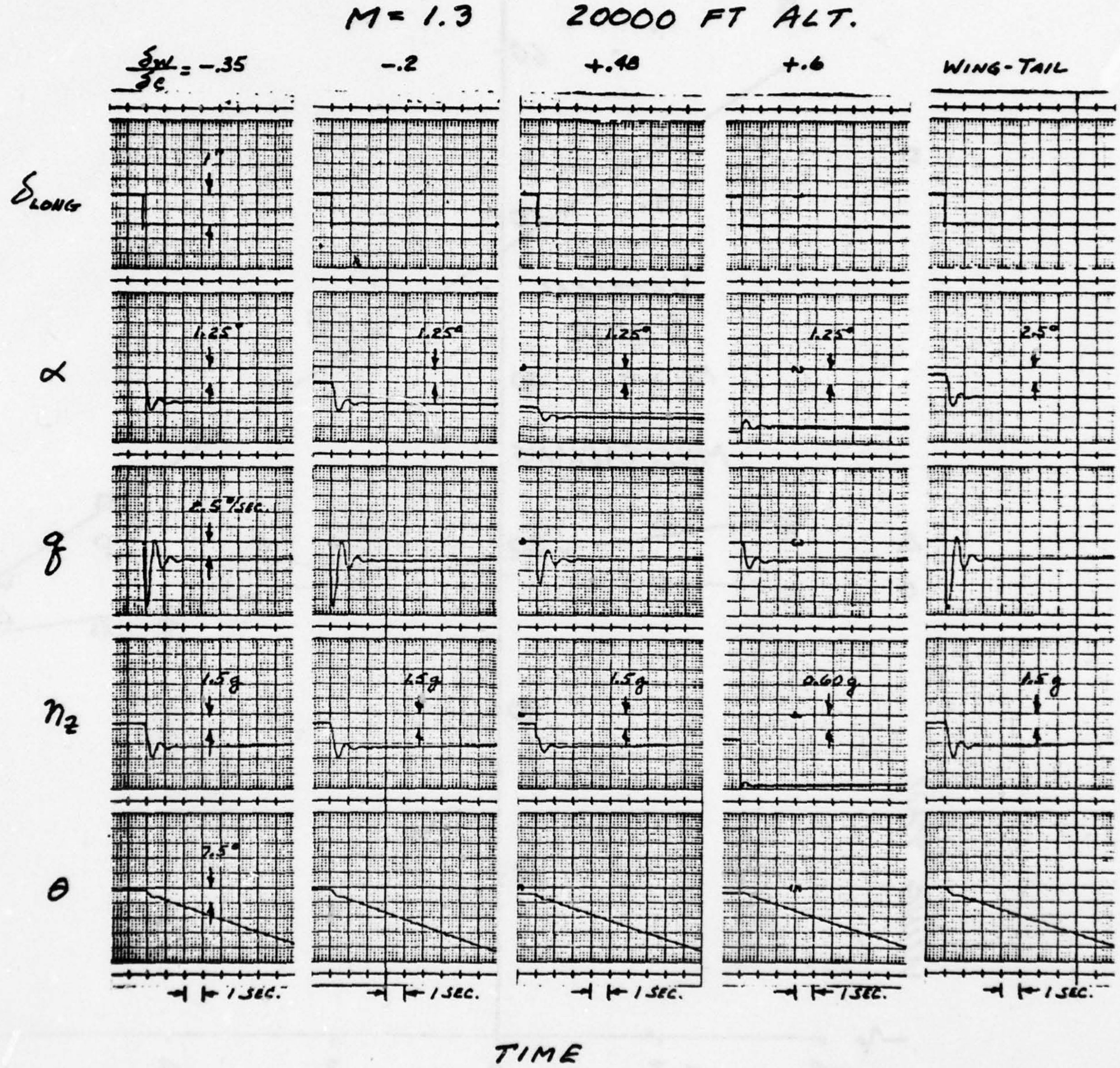

Figure 9c. Time Histories of Response to a Unit Step Stick Input

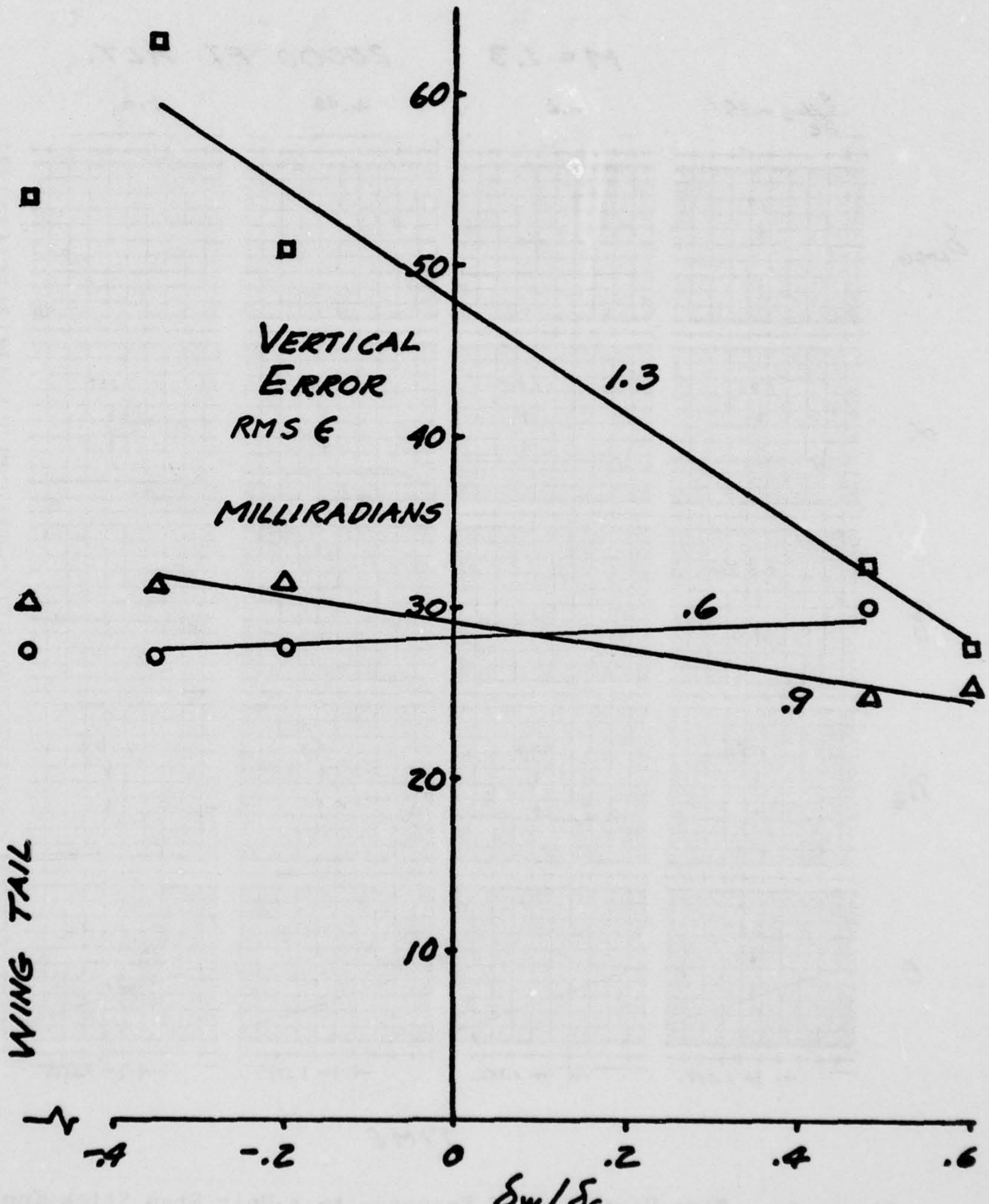

Figure 10. Vertical Tracking Error Results from Simulation

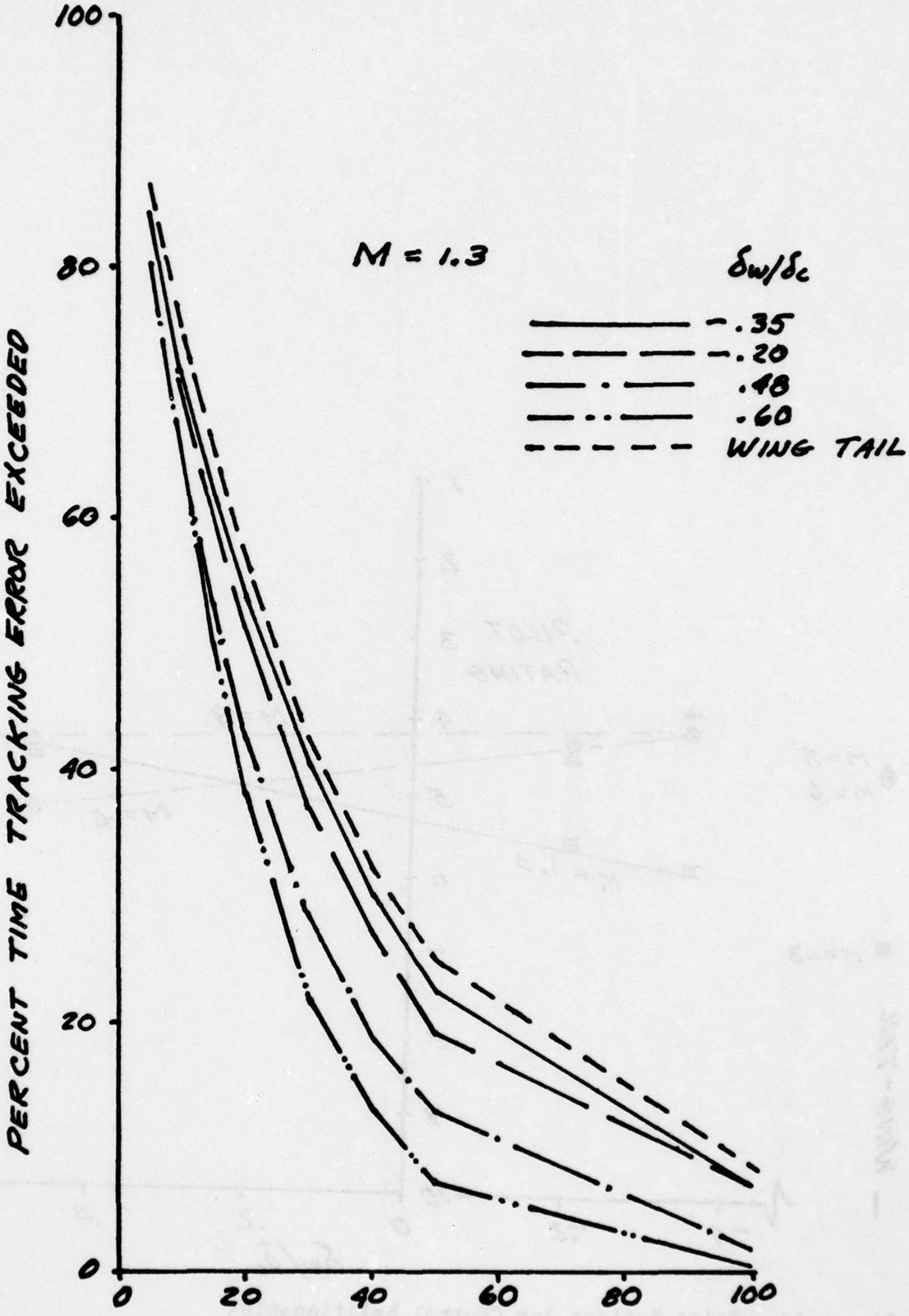

Figure 11. Typical Tracking Error Distributions

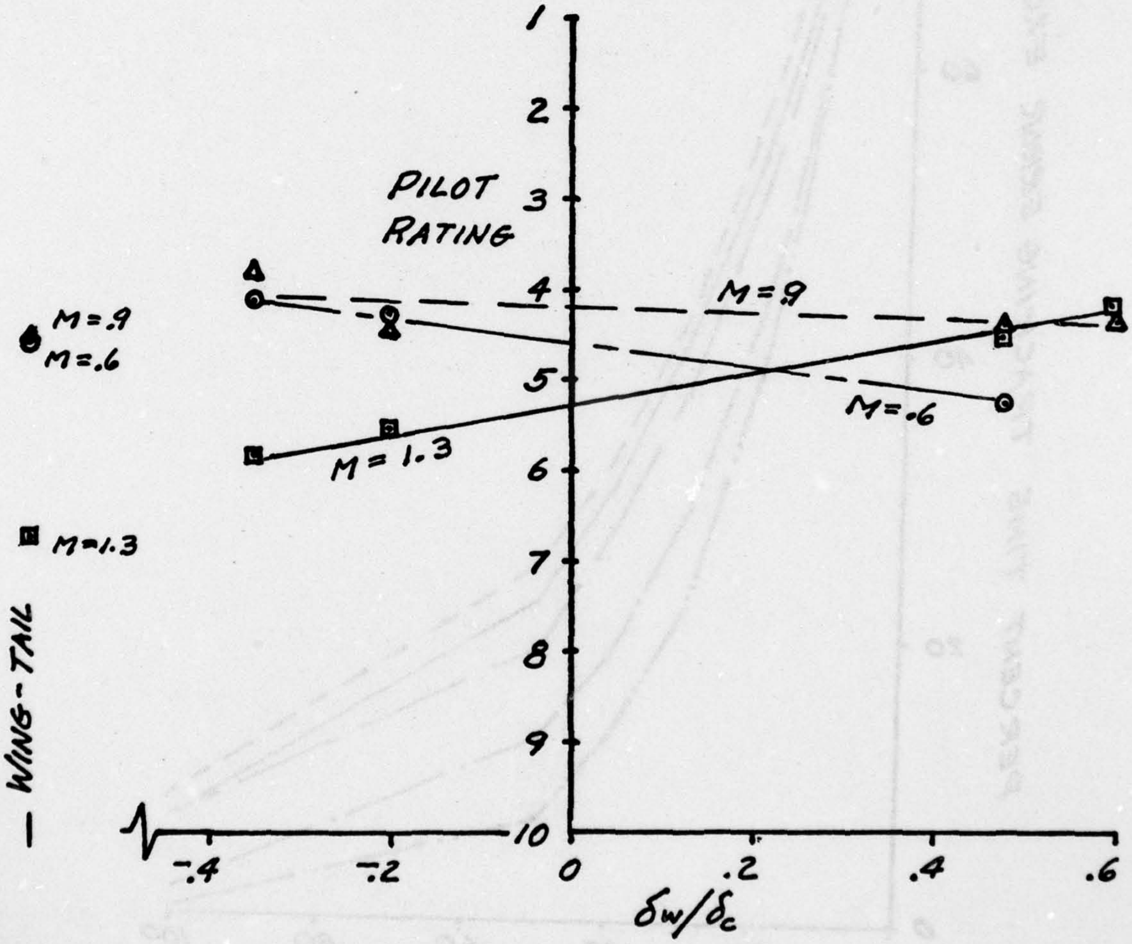

Figure 12. Pilot Ratings for Control Relationships

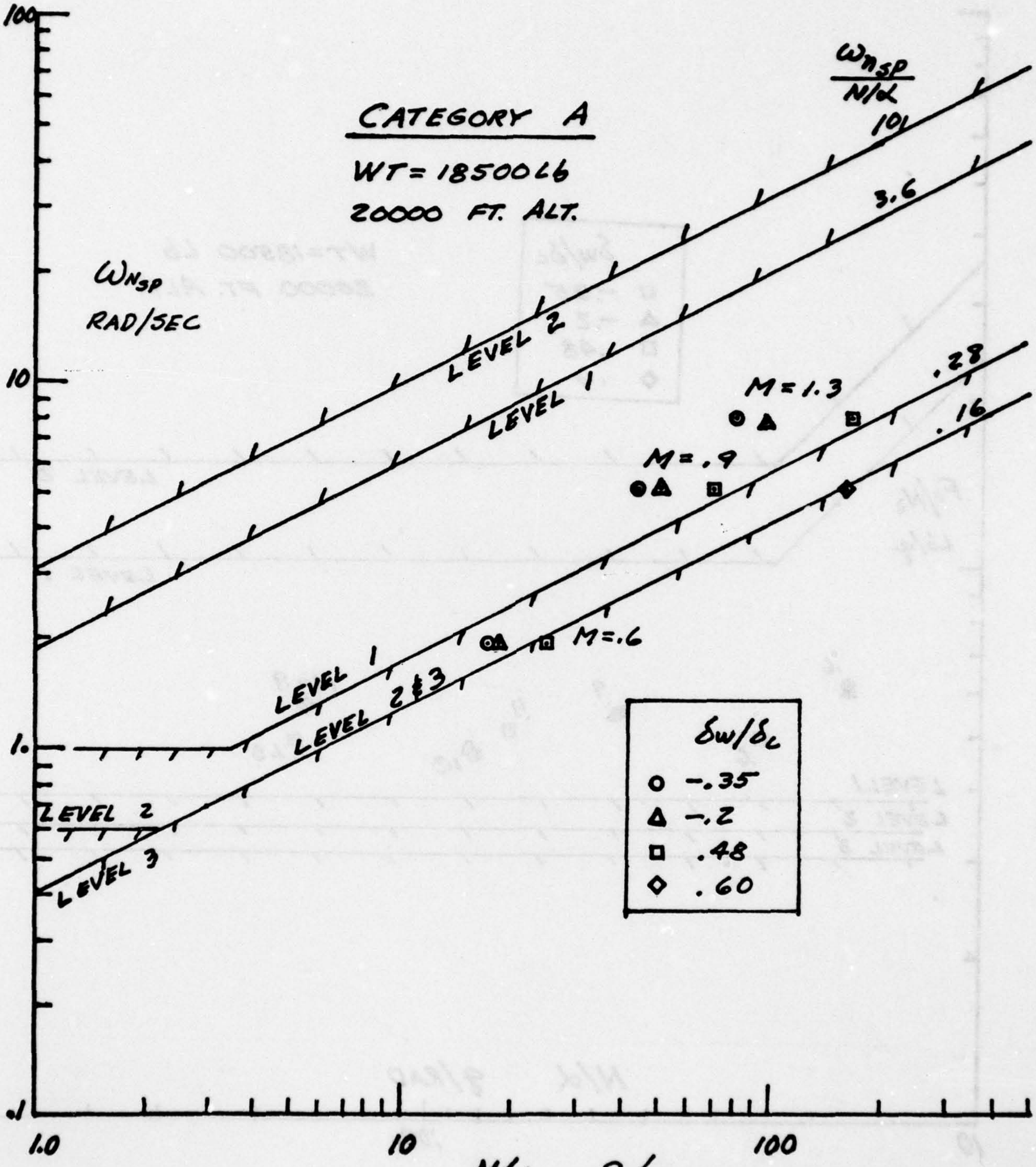

Figure 13. Short Period Frequency Requirements Comparison

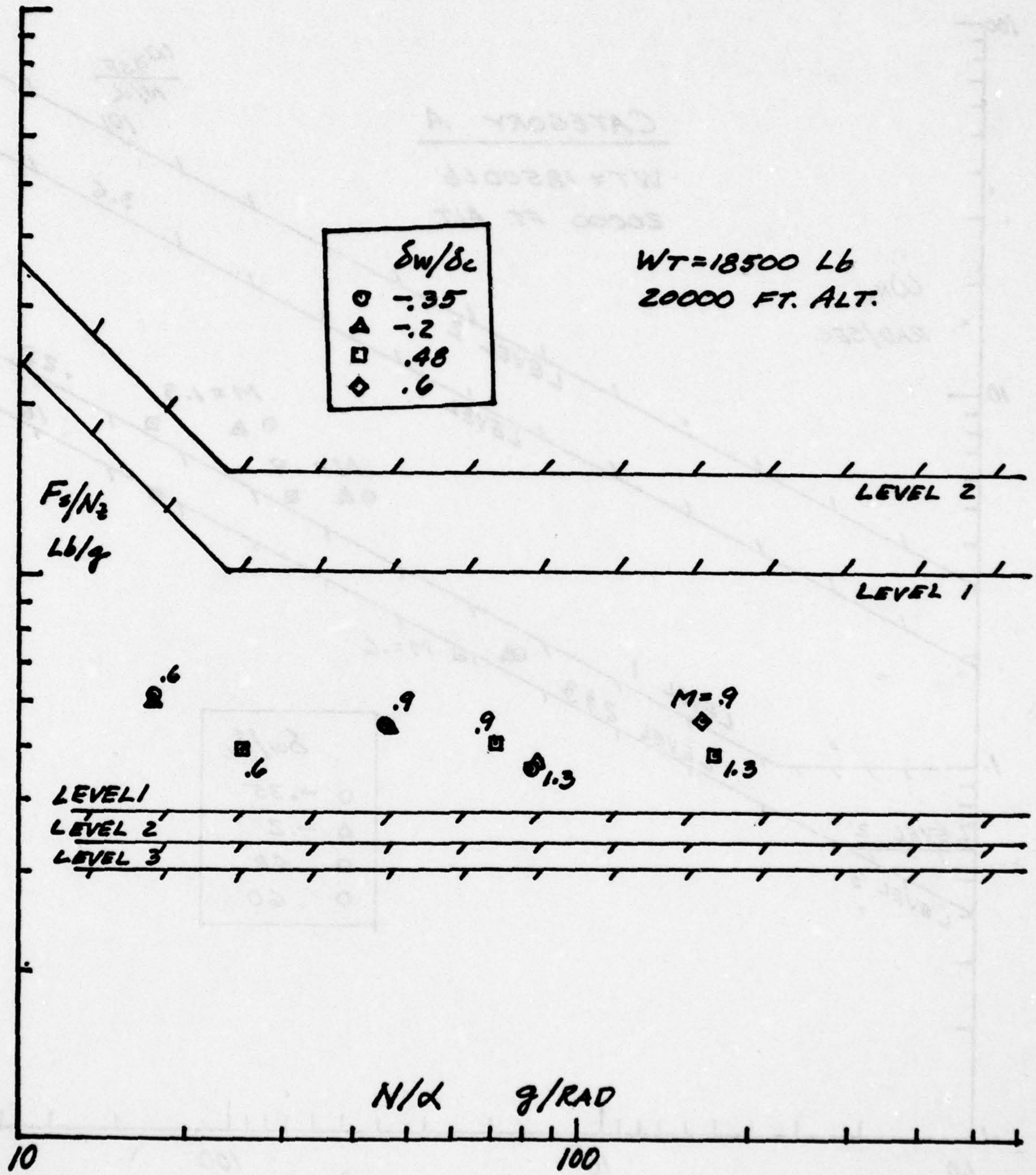

Figure 14. Maneuvering Force Gradient Requirements Comparison

Tom Twisdale, AFFTC: What maneuvers were used for tracking?
Answer: Large amplitude roll and pitching maneuvers. Rolls were about $\pm 100°$ and pitch maneuvers were a maximum of about +5, -1 g's. Abrupt motions were included also.

Jerry Lockenour, Northrop: The ω_n^2 vs n_z criterion of MIL-F-8785B has its origin in the CAP parameter ($\ddot{\theta}_{t=0} / n_{z_{ss}}$) which has required minimum and maximum levels. This is approximately a ratio of initial g's at the pilots station to steady state g's. Had we originally stated the requirement in this more fundamental form there would be no problem in analyzing higher order system and might be satisfactory even for direct lift control modes.
Answer: I'm not criticizing the parameter n_z/α. I think it's a fine parameter for evaluating typical airplanes. I'm just saying for direct lift control systems (longitudinal maneuvering) it's inappropriate and we need another kind of parameter to assess direct lift controls.

CALSPAN ADVANCED TECHNOLOGY CENTER

31 August 1978

ANOTHER STUDY OF THE T-38A PIO INCIDENT

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Staff Engineer

CRC/jc

A DIVISION OF CALSPAN CORPORATION
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ANOTHER STUDY OF THE T-38A PIO INCIDENT

INTRODUCTION

The T-38A aircraft as configured in the early 1960 versions was PIO prone at low altitude and high speed. A specific PIO incident was recorded during a flight of an instrumented airplane in 1960. Time histories of several parameters recorded during that incident have been published in various reports, References 1, 2, 3, and 4. These time histories are reproduced as Figure 1 in this memo. The T-38A PIO problem was studied by an Air Force review board in Reference 1, by Northrop in Reference 2, and by STI in Reference 3 during the time in 1963-64 when a fix for the airplane was being developed. The T-38A PIO incident has also been used as an example by other authors in attempts to develop theoretical explanations for that PIO incident and in attempts to develop requirements for incorporation in the flying qualities specification. Examples are the work by Neal in Reference 5 and Smith in Reference 4.

While reviewing the work reported by Smith in Reference 4, the author of this paper became interested in attempting to model the pilot's contribution to the dynamic system that would be required to cause the dynamic instability exhibited in the time histories of Figure 1. This paper documents the results of the study performed.

THE T-38A PIO TIME HISTORY

The following observations are drawn from study of the time histories of Figure 1.

1. At the beginning of the record, the flight control system is in a limit cycle oscillation with a frequency of $\omega = 22.2$ rad/sec. The pilot is not applying any force to the control stick.

2. At 7.7 sec on the time scale, the pilot switched the pitch damper OFF. This action interrupted the limit cycle with the stabilizer at one extreme of the oscillation. This trailing-edge-up stabilizer deflection caused the airplane to pitch up and pull 3 g.

3. During the first 0.3 sec after the pilot switched the pitch damper OFF, the pilot did not apply any stick force, but the stabilizer moved toward trim. This stabilizer motion could be a result of the action of the bobweight or it might be a result of the pitch damper actuator going through a "center and lock action".

4. At approximately 0.3 sec after the pilot switched the damper OFF, he applied a sharp push force to the stick to stop the pitch-up. At approximately $t = 8.4$ sec, the stick force is reversed briefly and for the next 5 seconds, the pilot-control system-airplane combination is involved in a violent PIO oscillation that is neutrally stable to slightly unstable. During this time, the mean pitch attitude is approximately 15^o nose up.

5. During the period $13 < t < 16$ sec, the pilot applied an average push force which caused the airplane to pitch over to approximately 7^o nose-up attitude.

6. During the period $16 < t < 21$ sec, the record exhibits a continuing oscillation of approximately 1/5 the previous amplitude.

7. The frequency of the PIO is approximately $\omega = 7.4$ rad/sec. This oscillation is exhibited in all of the recorded parameters. The stick force time history exhibits a higher frequency oscillation, $\omega \cong 19.6$ rad/sec, that is associated

with the control system. This oscillation is also very lightly damped.

8. The frequency and damping ratio of the oscillatory modes can be fairly accurately estimated from the time histories but relative phase information and amplitude information must be viewed with caution becuase the filtering of the sensors and recording channels is not known. Also, the pitch rate and normal acceleration traces are somewhat distorted sine waves which might indicate a structural mode is involved. The accelerometer trace is very smooth and exhibits no high frequency modes or "hash" typical of accelerometers. This could be the result of eyeball fairing the time history while hand tracing it for presentation in a report. The peaks of the $\dot{\theta}$ trace lag the maximum slopes of the θ trace by approximately 0.10 sec or 41^{o} at the PIO frequency. The stick force trace also exhibits a phase relative to the pitch attitude trace that is inconsistent with the transfer functions listed in Reference 6 for the control system, surface actuator and airplane. Evaluation of the θ/F_s transfer function at ω= 7.4 rad/sec gives

$$\theta/F_s = .404\, e^{-74.49^{o} j} \quad \text{deg/lb}$$

whereas estimation from the time histories indicates

$$\theta/F_s = .55\, e^{-255^{o} j} \quad \text{deg/lb}$$

Figure 2 illustrates the θ and F_s time histories and phase relation at ω = 7.4 rad/sec. This discrepancy between the calculated and measured values could be caused by the transfer functions not representing the airplane and control system. Since the phase shift is nearly 180^{o}, it suggests that possibly the push and pull labels on the F_s time history

may be reversed. Examination of the initial pilot input and its effect in stopping the pitch up and also the fact that the average push force applied from t = 13 to t = 16 sec did result in a nose-down airplane response establishes that the recording senses and the trace motions are consistent for low frequency. The values of the transfer function parameters used to calculate the amplitude ratio and phase angle at ω = 7.4 rad/sec were taken from Reference 6 which is bound as a section in Reference 1. The data used by STI was obtained from Northrop and Northrop derived the data from analysis, wind tunnel data, ground test of the control system and flight test on the T-38A family of aircraft. T-38A SN 59-1602 was the test aircraft used to develop fixes for the PIO problem. The PIO record illustrated in Figure 1 was taken in a different aircraft, SN 58-1194.

The large difference between the calculated and measured values of the phase of the θ response to F_s at ω = 7.4 rad/sec is of concern and must be kept in mind as a tempering factor in interpreting any analysis based on use of the transfer functions of Reference 6 to describe the T-38A dynamic characteristics. The stick-free transfer functions were developed by STI using pitch attitude to stabilizer deflection transfer function and stabilizer deflection to stick force transfer function data and calculation of the effect of normal acceleration at the bobweight fed back through the bobweight system. This analysis did not account for fore and aft acceleration effects on the control system or for angular acceleration effects on the control system components with significant rotary inertia.

9. Neither the stablizer nor the stick force trace exhibit any discontinuities that would indicate significant control system friction effects.

ROOT LOCUS ANALYSIS

The following assumptions were made as a basis for performing a root locus analysis of the T-38A PIO.

1. It was assumed that the transfer functions developed in Reference 6 represent the stick free T-38A with the bobweight active. See Table 1.

2. It was assumed that the system was linear. No account was taken of friction, etc.

3. It was assumed that the pilot-control system-airplane dynamic system during the PIO of Figure 1 is exhibiting two neutrally stable roots with frequencies

$$\omega_L = 7.4 \text{ rad/sec}$$

$$\omega_H = 19.6 \text{ rad/sec}$$

4. Because the stability of the high frequency flight control system mode is of interest and pilot transfer functions with transport delay are to be assumed, it was decided to use the exact representation of the transport delay developed in Reference 7 rather than the Pade approximation.

The T-38A PIO has been analyzed in Reference 3 and 4 using the following pilot transfer functions for pitch attitude loop closure and separately for loop closure of normal acceleration at the pilot station.

"Equalized" Pilot $\quad Y_{p_\theta} = \dfrac{K_\theta e^{-.2s}}{\left[\dfrac{s}{3.2} + 1\right]}, \quad Y_{p_{n_z}} = \dfrac{K_{n_z} e^{-.2s}}{\left[\dfrac{s}{3.2} + 1\right]}$

"Primitive" Pilot $\quad Y_{p_\theta} = K_\theta e^{-.2s}, \quad Y_{p_{n_z}} = K_{n_z} e^{-.2s}$

Root locus diagrams illustrating the effect on the closed loop system roots for each of these pilot transfer functions are contained in Figures 3, 4, 5 and 6. None of these root loci result in roots becoming unstable at the frequencies observed in the PIO time history, i.e., $\omega = 7.4$ and $\omega = 19.6$ rad/sec. Also none of these loci show two roots becoming unstable simultaneously at these two frequencies.

Many feedback parameters and forms for the pilot transfer function were assumed and root locus plots were calculated in attempts to force the short period and control system roots to become neutrally stable at $\omega = 7.4$ and $\omega = 19.6$ rad/sec for the same gain value. No solution was found for single variable feedback. Multiple solutions were found, however, when feedback of three airplane responses was used, e.g., θ, $\dot{\theta}$ and $\ddot{\theta}$ or n_z, θ and $\dot{\theta}$ with a time delay included in the pilot model.

The four pilot transfer functions listed in Table 2 will all cause the system root locus branches to pass into the right half plane at $\omega = 7.4$ and $\omega = 19.6$ for a given value of loop gain in each case. The root locus diagrams are shown in Figures 7-10.

Although these pilot model transfer functions are impressively complicated and the root locus diagrams are intricate and computer plotted, they are, in fact, meaningless because they are based on an invalid model of the T-38A airplane and control system. Once this fact was realized, the author set about trying to identify a more valid mathematical model of the T-38A and in the process "discovered" that the primary cause of the T-38A PIO phenomena was the horizontal stabilizer servo actuator control valve. The evidence available in the published reports that leads to this conclusion is presented and discussed below.

The discrepancy in phase angle of pitch attitude relative to stick force in the PIO record of Figure 1 relative to the phase angle calculated from the analytical model was noted previously and is illustrated in Figure 2. The two phase angles are different by 181°. In order to evaluate the

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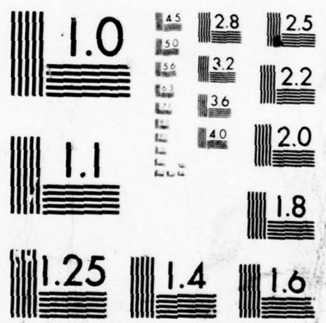

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NATIONAL BUREAU OF STANDARDS-1963-A

source of this discrepancy, a photo enlargement was made of Figure 1 and onion skin tracings were made of the traces. The traces could then be shifted and overlayed to establish fairly accurately their relative phase angles. The results of this analysis are shown in Figure 11 in the form of a phaser diagram using stick force as the reference. This figure illustrates the large difference in $\theta - F_s$ phase from the PIO record compared to the analytical model. It also illustrates a 41^o lag of pitch rate measured from the PIO record relative to a 90^o phase advance from the θ reference. This lag is probable due to a structural mode. This conclusion is supported by the shape of the $\dot{\theta}$ time history in Figure 12 (damper OFF). The n_z phaser in Figure 11 indicates some lag from the analytical phase lead from $\dot{\theta}$ that was calculated from the ratio of transfer function numerators. This lag might also be due to the structural mode or perhaps partly from recording filtering. The stabilizer time history in the PIO record goes off paper before the PIO sine wave becomes established and therefore its phase at the PIO frequency could not be estimated from the PIO record. The stabilizer phase relative to the pitch attitude phaser can be estimated from the stick fixed θ/δ_H transfer function, however, so effort was spent in searching out data to establish the stick-fixed and stick-free short period dynamics. The data in References 1, 2, 6, and 9 are plotted in Figure 13. The calculated values for stick-fixed and stick-free short period do not agree with the measured data. It seems apparent that the stick-free short period calculated in Reference 6 by assuming a simple n_z bobweight effect was not an adequate model. The stick-fixed data displayed in Figure 13 for doublet responses indicate the following short period roots:

$$\text{Stick Fixed} \quad \omega_{n_{sp}} = 7.0 \text{ rad/sec} \quad \zeta_{sp} = .4$$

The stick-free short period dynamics are less precisely defined by the data in Figure 13. The data indicate a stick-free model roughly as follows:

$$\text{Stick Free} \quad \omega'_{n_{sp}} = 7.0 \quad \zeta'_{sp} = .14$$

which is significantly different from the model in Table 1.

The stick-fixed θ/δ_H transfer function was evaluated at $\omega = 7.4$ which gave a phase lag of 128^o. This angle was used to locate the δ_H phaser relative to the θ phaser in Figure 11 for the PIO record set. These calculations suggest that the analytical model of the feel system and servo actuator δ_H/F_s is wrong by approximately 174^o. In Reference 5 the following stick-free transfer function for the T-38A control system is derived:

$$\frac{\delta_H}{F_s} = \frac{K\left[\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right]}{\left(\frac{s}{\tau_s'} + 1\right)\left[\frac{s^2}{\omega_{n_{F_s}}'^2} + \frac{2\zeta_{F_s}'}{\omega_{n_{F_s}}'} s + 1\right]\left[\frac{s^2}{\omega_{n_{sp}}'^2} + \frac{2\zeta_{sp}'}{\omega_{n_{sp}}'} s + 1\right]}$$

The phase contributions of each of the factors of this transfer function are noted on Figure 11 for the analytical model and for the model being derived from the PIO record and flight test data.

The reduction in the stick-free short period frequency from $\omega_{n_{sp}}' = 9.8$ to $\omega_{n_{sp}}' = 7.0$ rad/sec accounts for approximately 92^o of the 174^o difference in the δ_H/F_s phase angles shown in Figure 11. The next part of the transfer function to be reviewed was the stick-free feel system root. The Northrop reports state that the natural frequency of the feel system is primarily a function of the control system mass and feel spring. The natural frequency is appreciably higher than the PIO frequency, i.e., $\omega_{n_{F_s}} \approx 18$ rad/sec compared to 7.4 rad/sec so the phase shift contribution at 7.4 rad/sec is small. An adjustment to the feel system dynamics suggested in Reference 6 to account for the pilot's arm mass would reduce the natural frequency and damping ratio of the feel system to approximately $\omega_{n_{F_s}}' = 13$ rad/sec and $\delta_{F_s} = 0.13$. These values were used in the δ_H/F_s transfer function (PIO record) on Figure 11 to estimate the phase contribution of the feel system. Using the experimental estimates of the stick-fixed and stick-free short period, the feel system and the δ_H/F_s phase angle from the PIO record, the phase contribution of the horizontal stabilizer servo is established as approximately 101^o at $\omega = 7.4$ rad/sec.

The first order analytical model in Table 1 for the servo is (s =-21.7) which gives 18.7^o phase lag at ω = 7.4 rad/sec. Obviously, the servo can not be described as a first order linear element if it exhibits more than 90^o of phase shift. It is probable that the servo is higher order and possibly non-linear.

With the above analysis pointing to the horizontal tail servo as a major contributor of phase lag, a search was made of reports for commentary, data, time histories, frequency responses, etc. relating to the servo. The transient response in Figure 12 for damper OFF was photographically enlarged and the stick deflection doublet was transformed, through the nonlinear gearing curve of Figure 14, into a horizontal tail command δ_{H_c} and plotted in Figure 15. The actual horizontal tail response is also plotted on Figure 15. This comparison shows large attenuation and phase shift for an input that has frequency content near 12 rad/sec. This transient response verifies that the servo is not a 21.8 rad/sec linear first order element. Note that in performing the doublet input, the pilot applied full aft stick commanding 15^o of stabilizer deflection at a flight condition where the static load factor sensitivity is 2.32 g/deg. The horizontal stabilizer servo command and the horizontal stabilizer response to a similar doublet stick input is shown in Figure 15a for a test performed at high altitude and lower speed (i.e. lower dynamic pressure). This record indicates much less attenuation and smaller phase shift than the record taken at high dynamic pressure illustrated in Figure 15. These two responses suggest that the servo response is a function of hinge moments. Examination of the PIO time history in Figure 1 between 8-8.5 sec shows that the pilot applied a push-pull doublet of 34 lb push and 24 lb pull. It should be noted, however, that the peaks in the force application are phased so that the pilot-applied force and the bobweight force add. Thus, the total force applied to the feel spring is 37 lb push and 31 lb pull. Statically, these force applications, see Figure 16, would command 14^o TED and 6^o TEU of horizontal tail deflection. The frequency content of the doublet is roughly 10-15 rad/sec. If the analytical model of the servo actuator were correct, this command would have been followed with unity gain and small phase lag. The PIO record, however, indicates only 4^o TED and 0.6^o TEU response with considerable phase shift. This record also indicates that the servo has not been modeled correctly.

The dynamic characteristics of the horizontal tail servo are discussed in Reference 8 and curves relating servo valve flow as a function of valve displacement are presented, in Figure 17, for three valves that were under study. The flow characteristics of the valve used on airplanes with the PIO problem is not included on Figure 17. A frequency response performed on the controls test stand for the servo valve used on the production airplane with the PIO problem (Configuration D) is shown in Figure 18 for two levels of force input, $F_s = \pm 7$ lb and $F_s = \pm 15$ lb. These two frequency responses exhibit nonlinear response with amplitude and considerable reduction in the value of the first order root in the third order model used to fit the test data. It is most probable that the servo valve flow is a function of load pressure as indicated in Figure 19. The parabolic curves in Figure 19 are typical of hydraulic flow control valves. When the load pressure becomes a large fraction of the supply pressure, the gradient of flow output with valve displacement is reduced and the dynamic response of the servo is degraded, however, the servo is capable of passing low frequency inputs and holding static loads as long as the load pressure does not exceed the system supply pressure. Reference 4 states on page 122 that the surface deflection rates are a function of the air load hinge moments.

Commentary and evaluation data from simulator experiments indicated that tests of a servo valve with time constant reduced by 1/2 resulted in "definitely reduced PIO tendencies with no degradation in flying qualities". The flight test program performed to evaluate various changes to the flight control system is described in Reference 1. Table 1 from that report documents the chronology of modifications evaluated. Note that the new servo valve was introduced in Mod I and by Mod IV most of the control system changes that were eventually adopted to fix the PIO problem had been introduced, i.e., trigradient spring, reduced bobweight, reduced preload, new servo valve. Only the change to the nonlinear gearing had not yet been made. In Mod VI the original servo valve was reinstalled. The evaluation comments were as follows (from page D-27 of Reference 1): "The pilot evaluated Mod VI no better than the baseline control system for both PIO and Cooper ratings. The control system performance deterioration was so distinct that only one flight was made with this configuration". This test indicates that the major cause of the PIO problem was the low dynamic performance of the servo.

The question might be asked — "Why spend time analyzing a problem that was adequately solved fifteen years ago?" The reason the study was performed was because the T-38A PIO incident has been used by a number of researchers to test their theories for what causes pilot-induced oscillations. In referring to the incident, however, they all use the analytical model developed in Reference 6 to represent the airplane. This model is incorrect and therefore analysis based on the analytical model or correlation of PIO theories with the parameters in the analytical model such as those in References 3, 4 and 5 are invalid.

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2. Hirsch, D. L.: "Investigation and Elimination of PIO Tendencies in the Northrop T-38A," paper presented at the SAE Meeting, July 1964, New York, Northrop Norair.

3. Ashkenas, I. L., Jex, H. R. and McRuer, D. T.: "Pilot-Induced Oscillations: Their Cause and Analysis," Norair Report NOR-64-143, Systems Technology, Inc. Report STI TR-239-2, 20 June 1964.

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5. Chalk, C. R., Neal, T. P., Harris, T. M. and Pritchard, F. E.: "Background Information and User Guide for MIL-F-8785A(ASG) "Military Specification - Flying Qualities of Piloted Airplanes," prepared for AFFDL, June 1969.

6. Jex, H. R.: "Summary of T-38A PIO Analyses," STI TM No. 239-3, 25 January 1963.

7. Schubert, G. R.: "Digital Computer Program for Root Locus Analyses of Open-Loop Transfer Functions Containing an $e^{-\tau s}$ Lead-Lag Term," AFFDL FDCC TM 65-43, September 1965.

8. Nelson, W. E.: "An Analytical Investigation for the Possible Optimization of the T-38A Longitudinal Control System," Northrop Norair Report No. NOR-64-52 (FMR-61-2), February 1964.

9. Lusby, W. A., Jr., and Hanks, N. J.: "T-38A Category II Stability and Control Tests", Report No. AFFTC-TR-61-15, August 1961.

Table 1

T-38A STICK-FREE TRANSFER FUNCTIONS

$$\frac{\theta}{F_s} = \frac{.0084\left[\frac{s}{3.18} + 1\right]}{s\left[\frac{s}{21.8} + 1\right]\left[\frac{s^2}{9.8^2} + \frac{2(.10)}{9.8}s + 1\right]\left[\frac{s^2}{17.7^2} + \frac{2(.23)}{17.7}s + 1\right]}$$

$$\left.\frac{n_z}{F_s}\right|_{\substack{K_B = 2.0 \\ F.S. = 200}} = \frac{.248\left[\frac{s^2}{24.4^2} + \frac{2(.17)}{24.4}s + 1\right]}{\left[\frac{s}{21.8} + 1\right]\left[\frac{s^2}{9.8^2} + \frac{2(.10)}{9.8}s + 1\right]\left[\frac{s^2}{17.7^2} + \frac{2(.23)}{17.7}s + 1\right]}$$

Flight Condition

$M = .85 \sim .90$ C.G. = 16%

h = S.L. $\sim$ 5,000 ft   W = 11,690 lb

F.S. = 200 is pilot station

Table 2

PILOT MODELS THAT CAUSE ROOT LOCI

TO BECOME UNSTABLE AT ω = 7.4 AND 19.6 RAD/SEC

NO. 1

$$Y_p = \left. \frac{184.2 \left[\frac{s^2}{12.54^2} + \frac{2(.03064)}{12.54} s + 1 \right]}{\left[\frac{s^2}{18.6^2} + \frac{2(.14)}{18.6} s + 1 \right]} e^{-.239s} \right|_{\omega = 7.4} = 141.4\, e^{-108.4^{\circ} j}$$

NO. 2

$$Y_p = \left. \frac{372.3 \left[\frac{s^2}{10.37^2} + \frac{2(.16)}{10.37} s + 1 \right]}{\left(\frac{s}{5.5} + 1 \right) \left[\frac{s^2}{18.6^2} + \frac{2(.14)}{18.6} s + 1 \right]} e^{-.164s} \right|_{\omega = 7.4} = 141.54\, e^{-105.5^{\circ} j}$$

NO. 3

$$Y_p = \left. \frac{11.45 \left(-\frac{s}{.540} + 1 \right) \left(\frac{s}{9.574} + 1 \right)}{\left(\frac{s}{5.5} + 1 \right) \left(\frac{s^2}{18.6^2} + \frac{2(.14)}{18.6} s + 1 \right)} e^{+.015s} \right|_{\omega = 7.4} = 139.7\, e^{-102.7^{\circ} j}$$

NO. 4

$$Y_p = \frac{K_{\theta} (.0084) \left[\frac{s^2}{11^2} + \frac{2(.32)}{11} s + 1 \right]}{\left[\frac{s^2}{18.6^2} + \frac{2(.14)}{18.6} s + 1 \right]} e^{-.0957s}$$

$\left[n_z ,\ \theta,\ \theta \text{ feedback} \right]$

TABLE I

Mod. No	CONTROL SYSTEM CHANGE	EOC No.	Test Flight No.
0	Baseline T-38A Control System A. Spring Gradient dF_s/dT_s=4.5 lb/in) See Fig. 3 Spring Preload F_p=1.25 lb. B. Total Bob-weight effectiveness dF_{BW}/dT_s=1.7 lb/g at δH=1° T.E. UP. See Fig. 6		162 thru 168 incl.
I	A. Increased stick force spring gradient-See Fig. 3 to dF_s/dT_s= 10 lb/in Flt. 169 Spring Preload F_p=0.5 lb. at stick grip Flt. 172 Spring Preload F_p=1.6 lb. at stick grip B. New Servo-Valves with reduced time constant 1/2 that of production servo-valves	74519 74270 69999	 169 thru 172 incl.
II	A. Stick force spring gradient changed-See Fig. 3 to dF_s/dT_s=7.0 lb/in Spring Preload F_p=0.8 lb. at stick grip B. Reduced Bob-Weight effectiveness to 1/2 T-38A 1. Modification of Bob-Weight Mechanism-See Fig 6 2. Removal of Aft Control Stick 3. Replace Bob-Wt. Balance Spring C. New Servo-Valves (Mod I)	74519 74540 74552 74548 Noted	 174 thru 177 incl.
III	A. Tri-Gradient Control Force Spring-See Fig. 4 Gradient #1 dF_s/dT_s= 10 lb/in #2 dF_s/dT_s= 8.5 lb/in #3 dF_s/dT_s= 7 lb/in Spring Preload F_p= 0.15 lb. at stick grip B. Reduced Bob-Weight Effect (Mod II-B) C. New Servo-Valves (Mod I-B) ?	74579 74593 74594 Noted Noted	 179 thru 181 incl.
IV	A. Tri-Gradient Spring-See Fig. 4 Gradient #1 dF_s/dT_s= 16 lb/in to F_{s1}= 1.5 lb #2 dF_s/dT_s= 11.8 lb/in to F_{s2}= 2.7 lb #3 dF_s/dT_s= 7.6 lb/in Spring Preload F_p= 0.5 lb. at stick grip B. Reduced Bob-Weight Effect (Mod II) C. New Servo-Valves (Mod I)	74580 75615 74579 75621 Noted Noted	 182 and 184 only
V	Same as Mod IV, except Removed Bob-Weight Balance Spring	75624	183 only
VI	Same as Mod IV, except Replaced Production Servo-valves (Bob-Wt. Balance Spring Re-Installed)	75624	185 only

TABLE I Control System Test Modifications on aircraft AF 60-602 (N5115) for PIO Flight Test Program.
Sheet 1 of 2

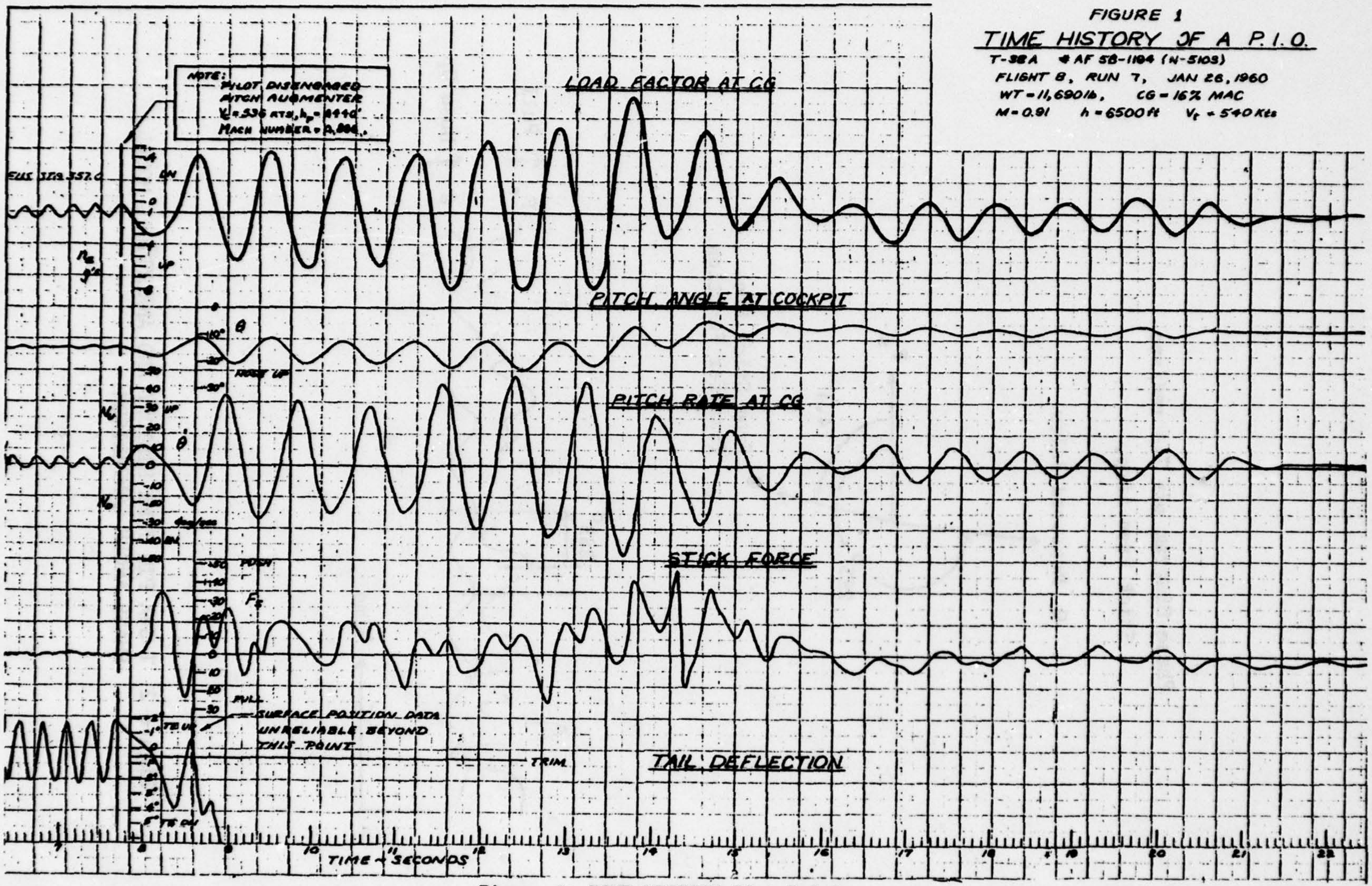


Figure 1 TIME HISTORY OF A P.I.O.

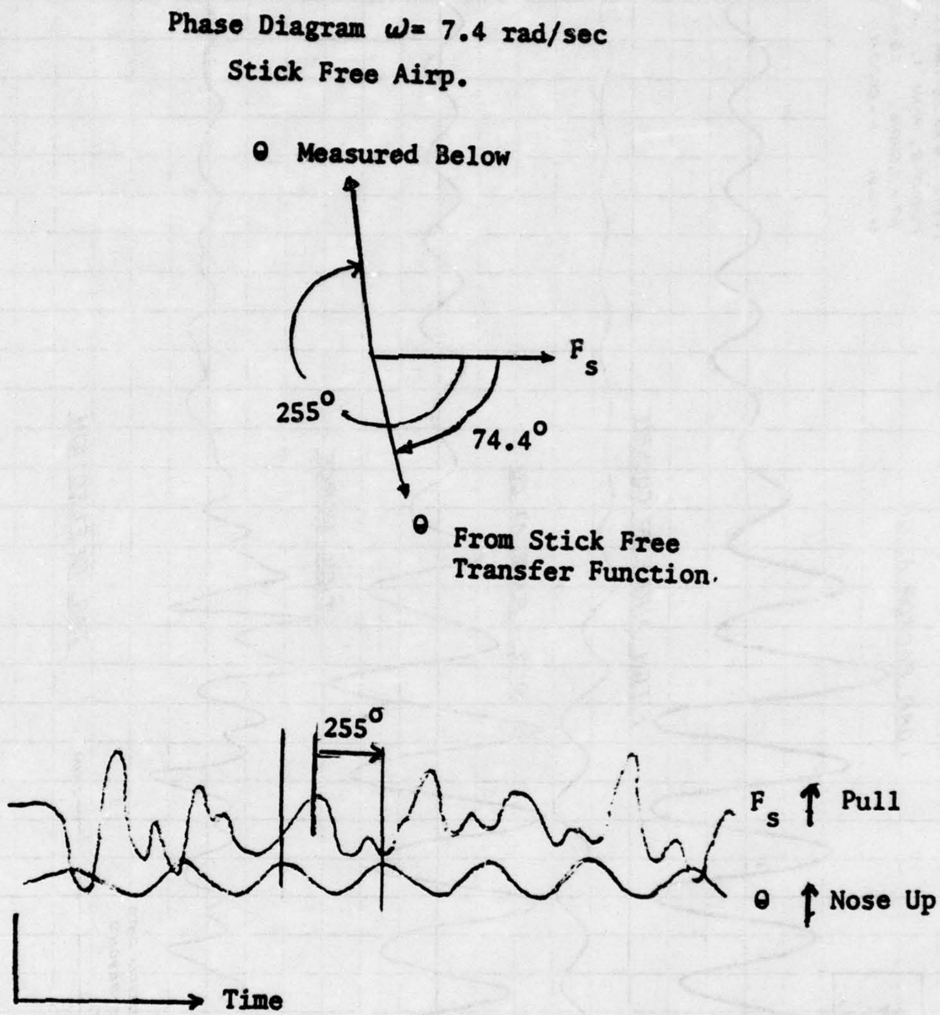


Figure 2 T-38A PIO TIME HISTORY AND ANALYTICAL MODEL PHASE ANGLE COMPARISON

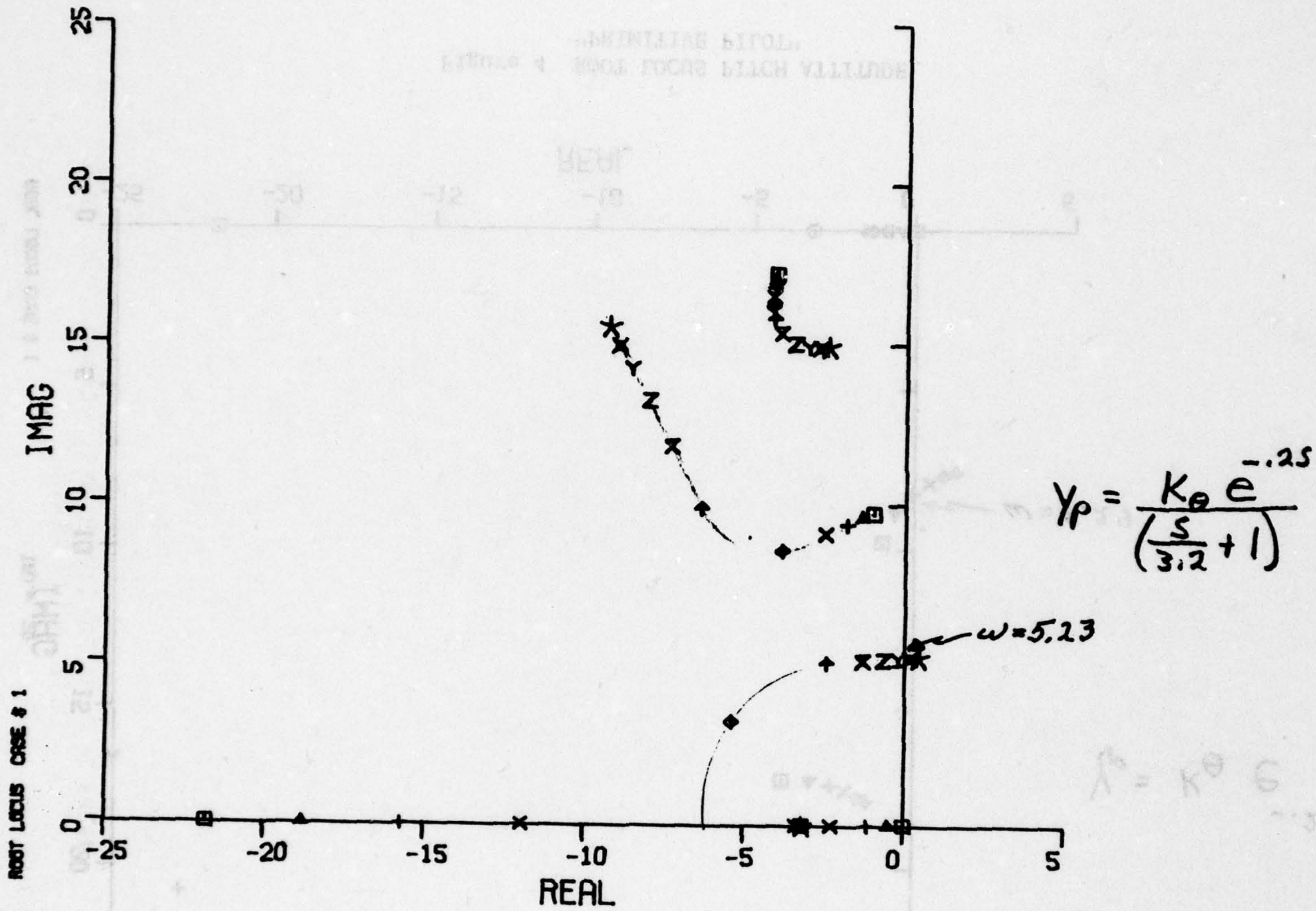


Figure 3 ROOT LOCUS PITCH ATTITUDE
"EQUALIZED PILOT"

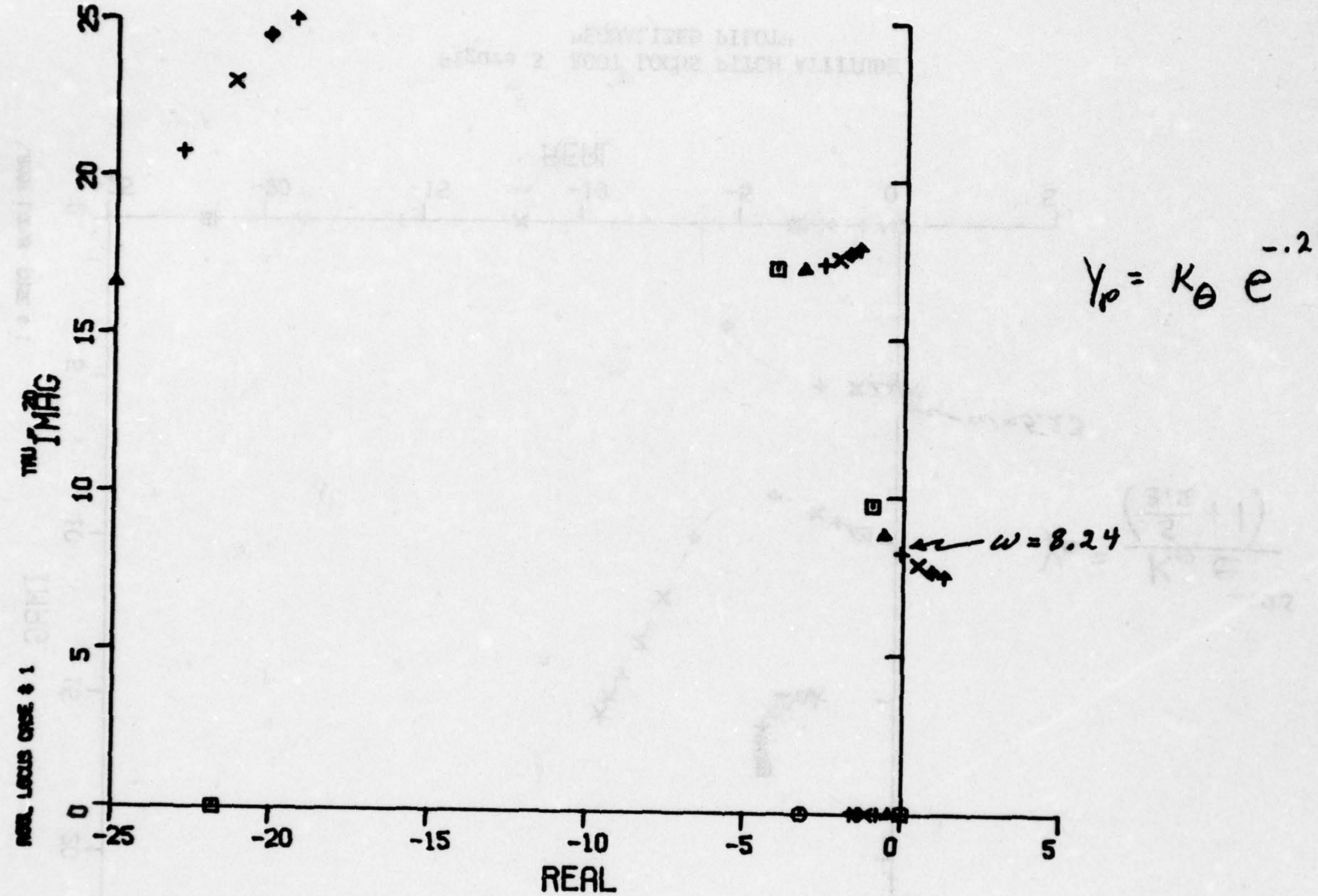


Figure 4 ROOT LOCUS PITCH ATTITUDE
"PRIMITIVE PILOT"

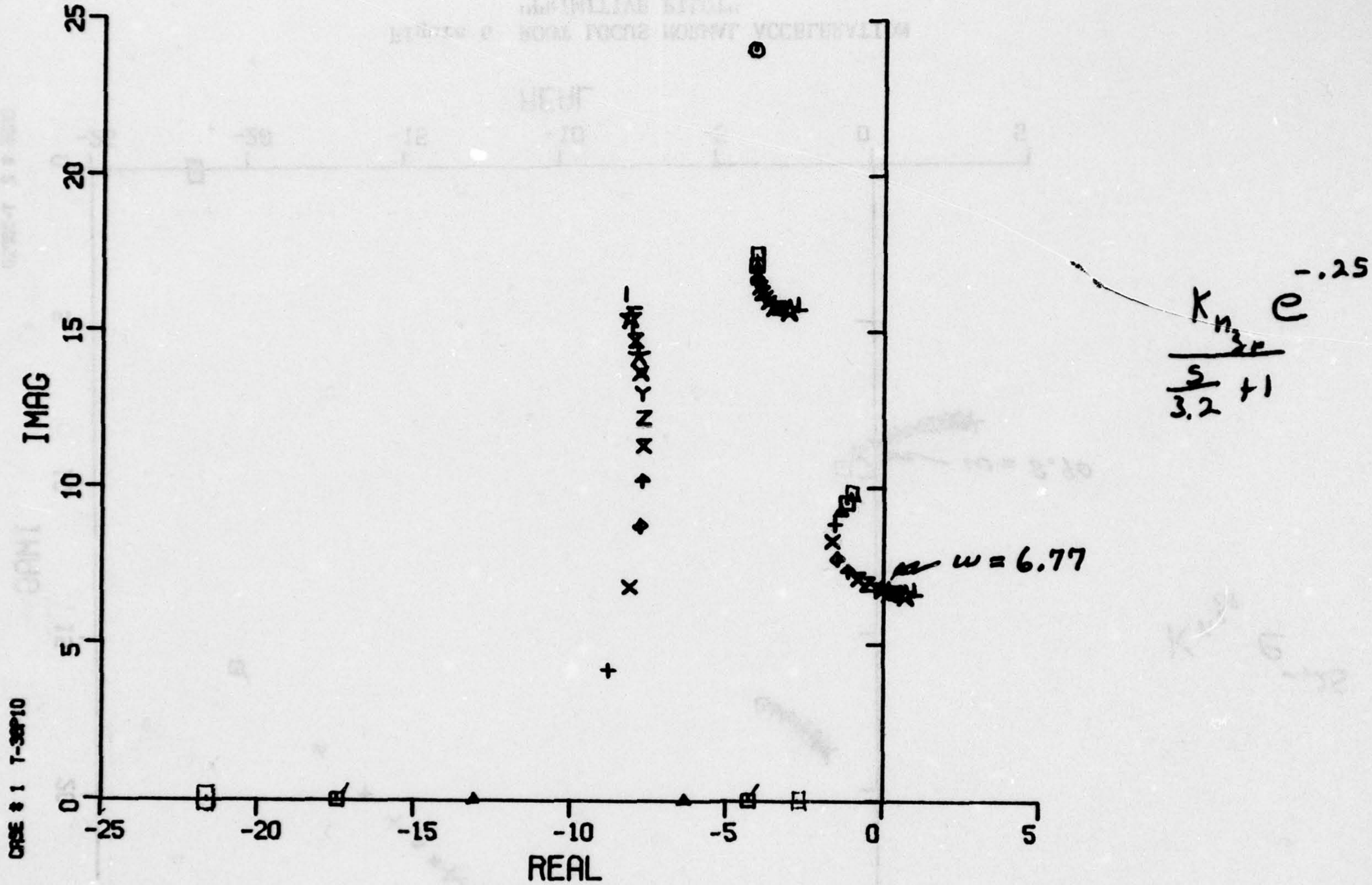
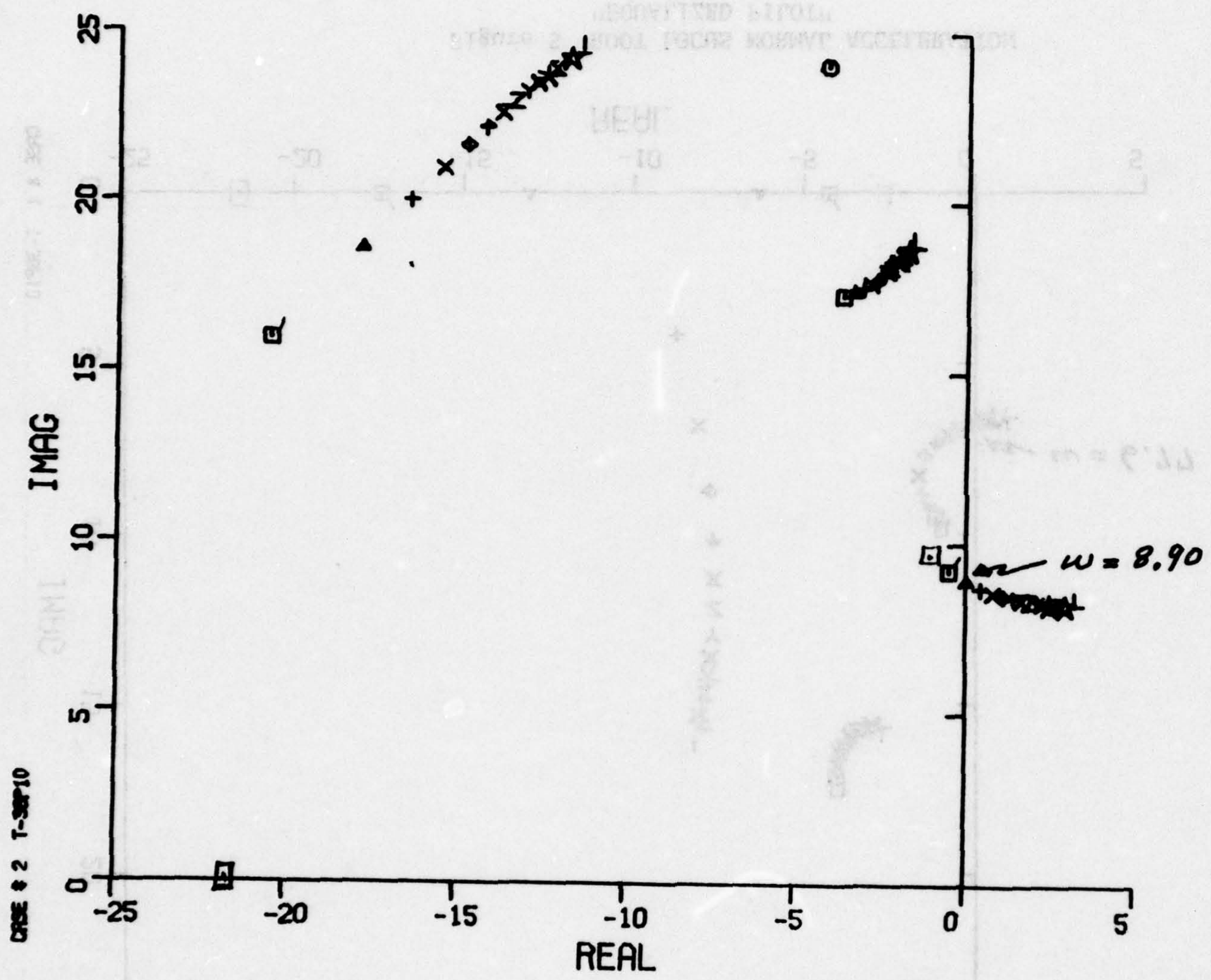
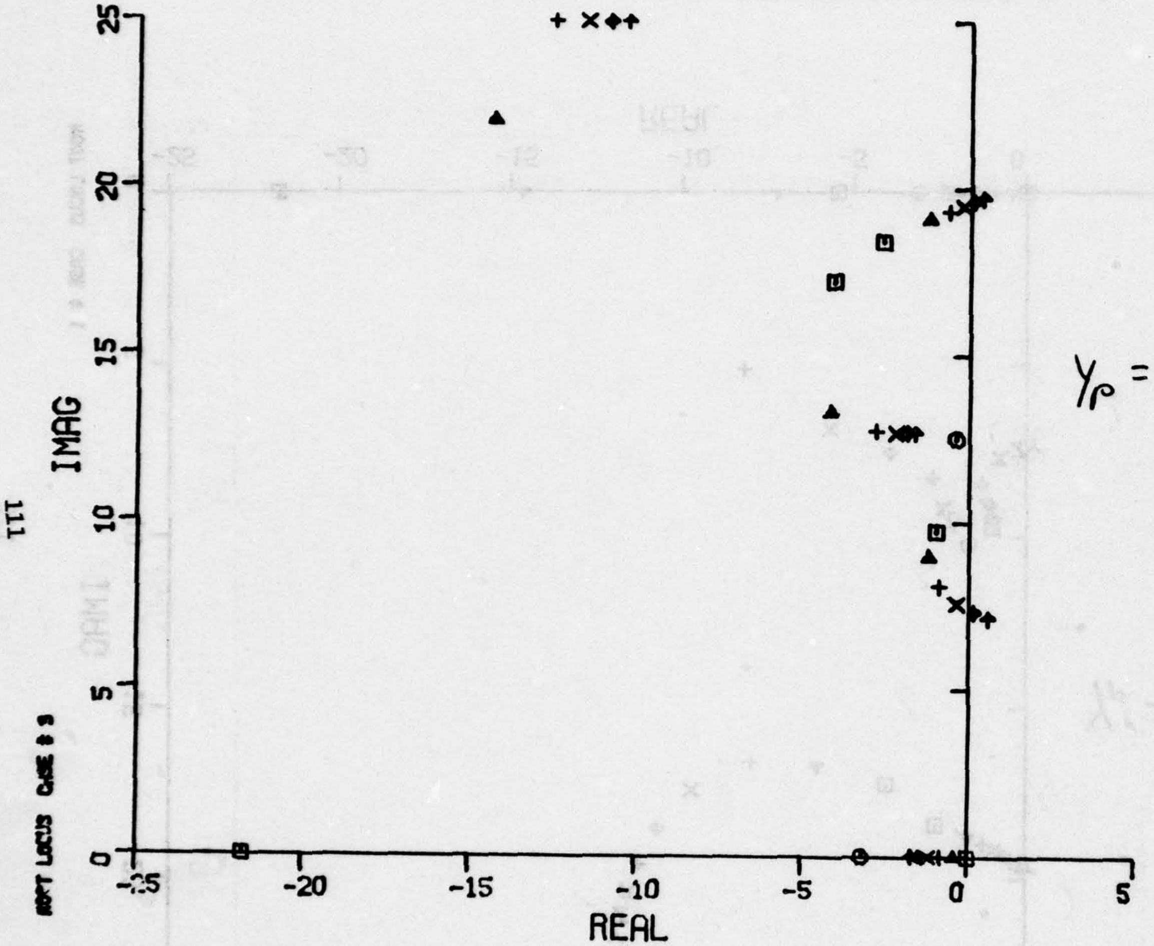


Figure 5 ROOT LOCUS NORMAL ACCELERATION
"EQUALIZED PILOT"



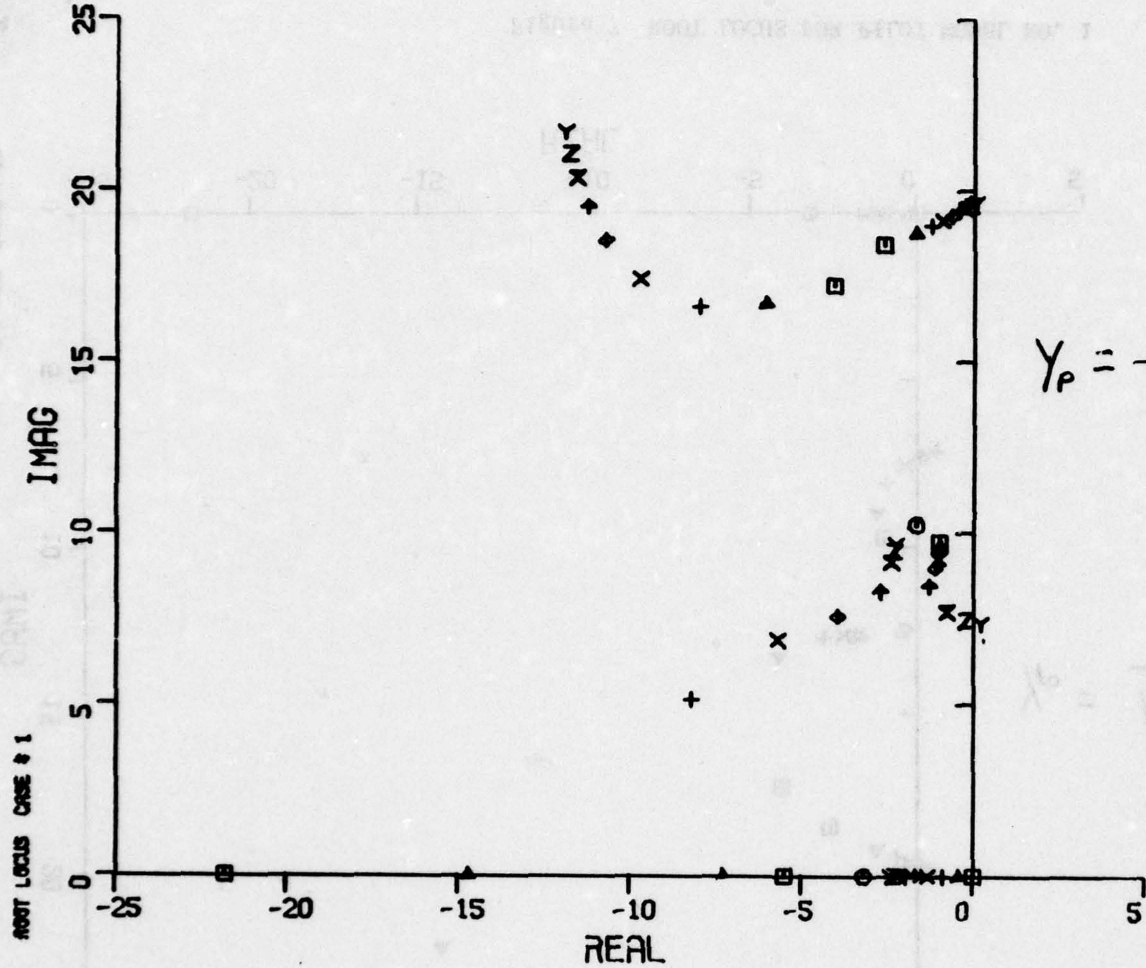
$K_{n_{z_{\delta p}}} e^{-.25}$

Figure 6 ROOT LOCUS NORMAL ACCELERATION "PRIMITIVE PILOT"



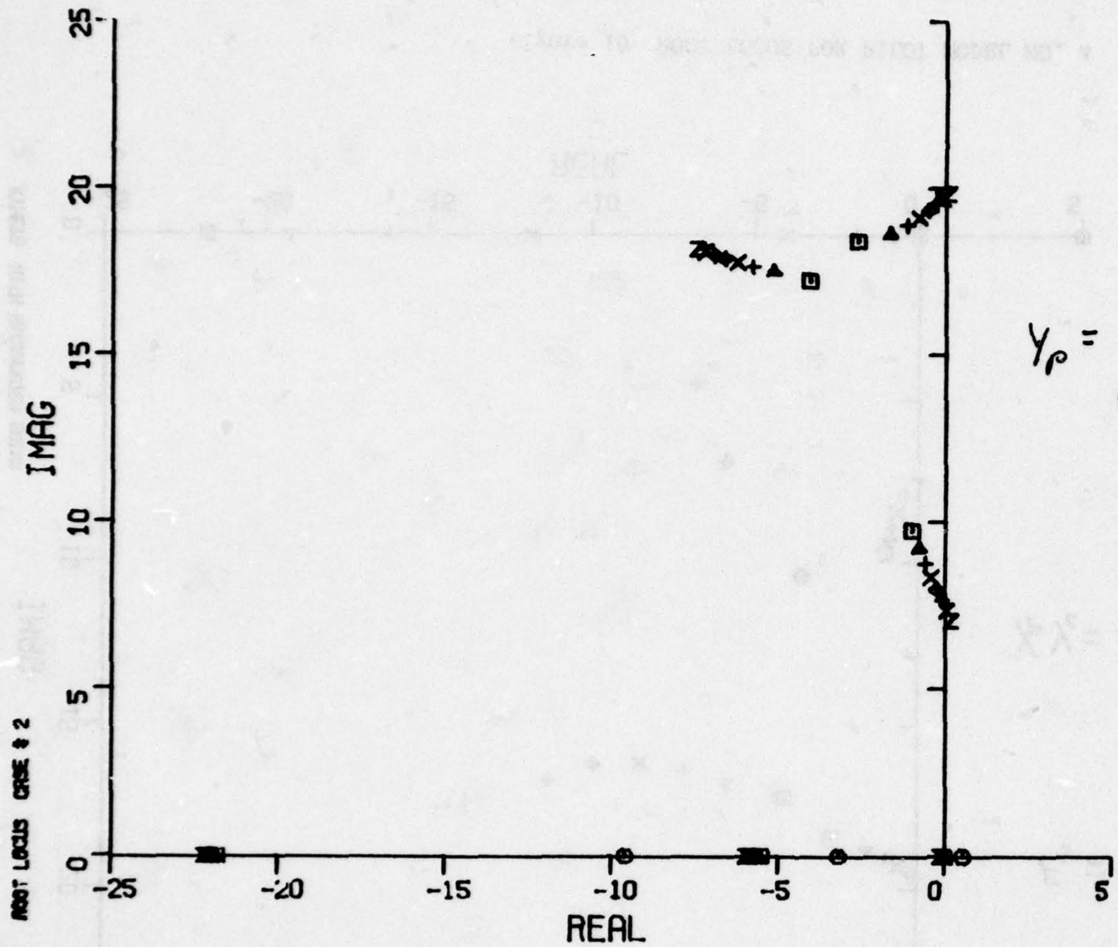
$$Y_p = \frac{K_\theta \left[\frac{s^2}{12.542^2} + \frac{2(.03064)}{12.542} s + 1\right] e^{-.239s}}{\left[\frac{s^2}{18.6^2} + \frac{2(.14)}{18.6} s + 1\right]}$$

Figure 7 ROOT LOCUS FOR PILOT MODEL NO. 1



$$Y_p = \frac{K_\Theta\left[\frac{s^2}{10.372^2} + \frac{2(.16)}{10.372}s + 1\right]e^{-.164s}}{\left(\frac{s}{5.5}+1\right)\left[\frac{s^2}{18.6^2} + \frac{2(.14)}{18.6}s + 1\right]}$$

Figure 8 ROOT LOCUS FOR PILOT MODEL NO. 2

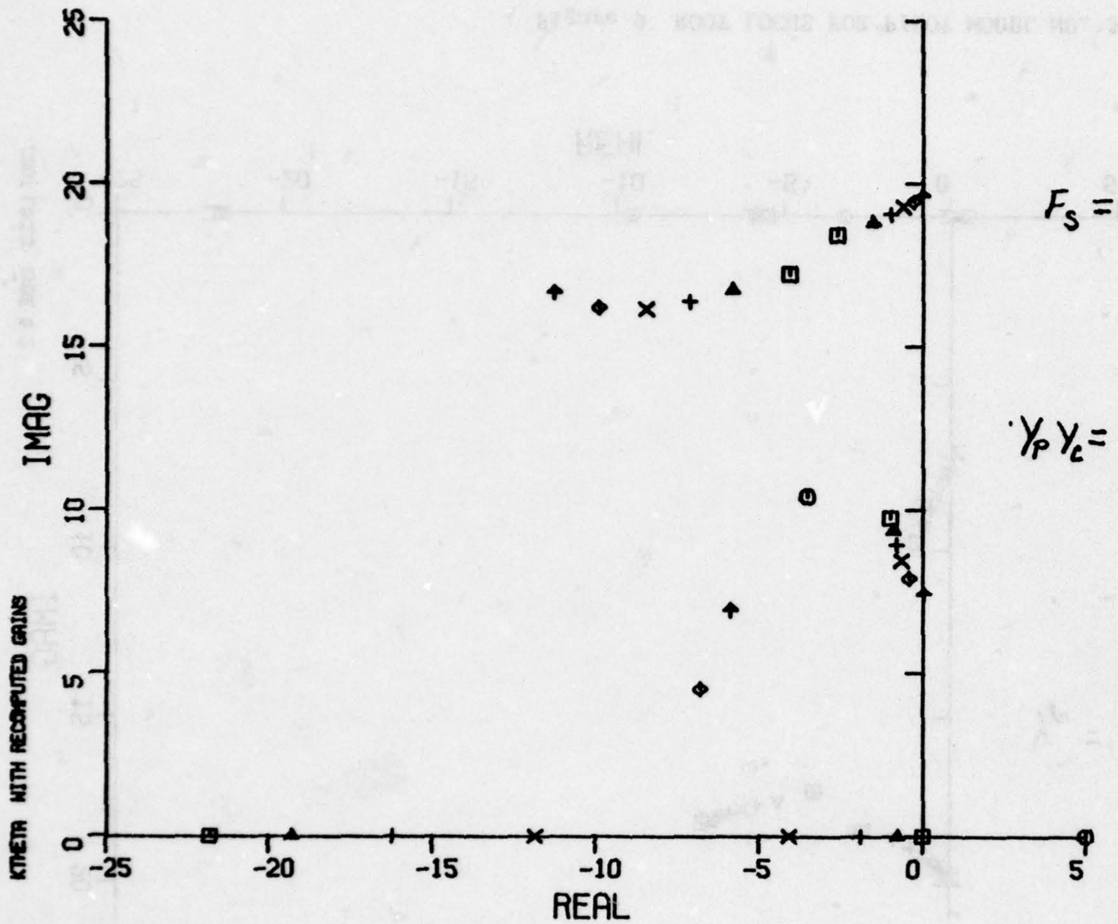


$$Y_p = \frac{K_\theta \left(\frac{s}{-0.540}+1\right)\left(\frac{s}{9.574}+1\right) e^{+.015}}{\left(\frac{s}{5.5}+1\right)\left[\frac{s^2}{18.6^2}+\frac{2(.14)}{18.6}s+1\right]}$$

Note

τ ~ Anticipation (small)

Figure 9 ROOT LOCUS FOR PILOT MODEL NO. 3



$$F_S = \frac{\left[K_{n_z} n_{z_\epsilon} + K_\theta \theta_\epsilon + K_{\dot\theta} \dot\theta_\epsilon\right] e^{-.2s}}{\left[\frac{s^2}{18.6^2} + \frac{2(.14)}{18.6} s + 1\right]}$$

$$Y_P Y_C = \frac{K_\theta (.0084)\left[\frac{s^2}{11^2} + \frac{2(.32)}{11} s + 1\right] e^{-.0957s}}{\left[\frac{s^2}{18.6^2} + \frac{2(.14)}{18.6} s + 1\right] \Delta_{T\text{-}38A}}$$

Figure 10 ROOT LOCUS FOR PILOT MODEL NO. 4

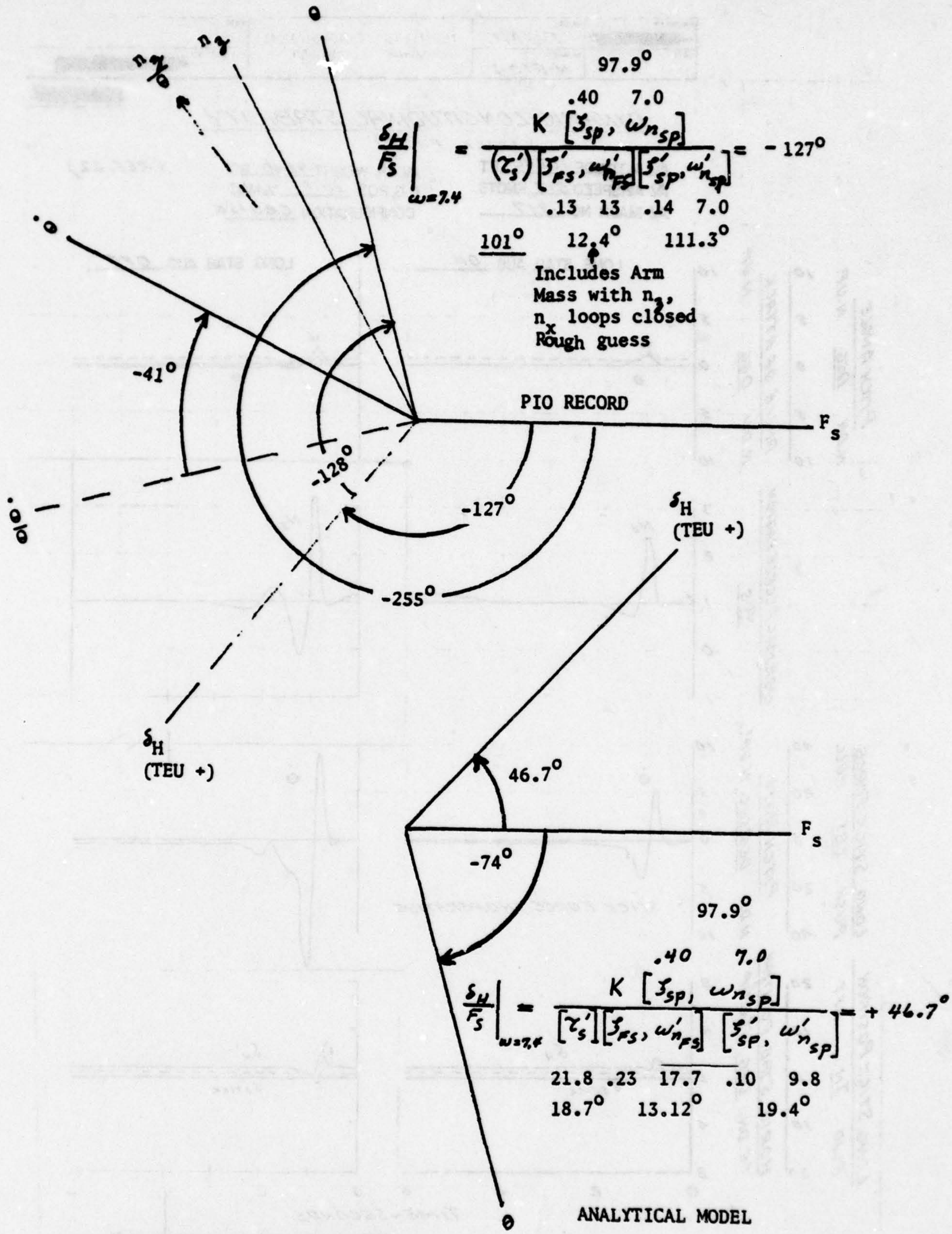


Figure 11 PHASE RELATION OF PIO TIME HISTORIES

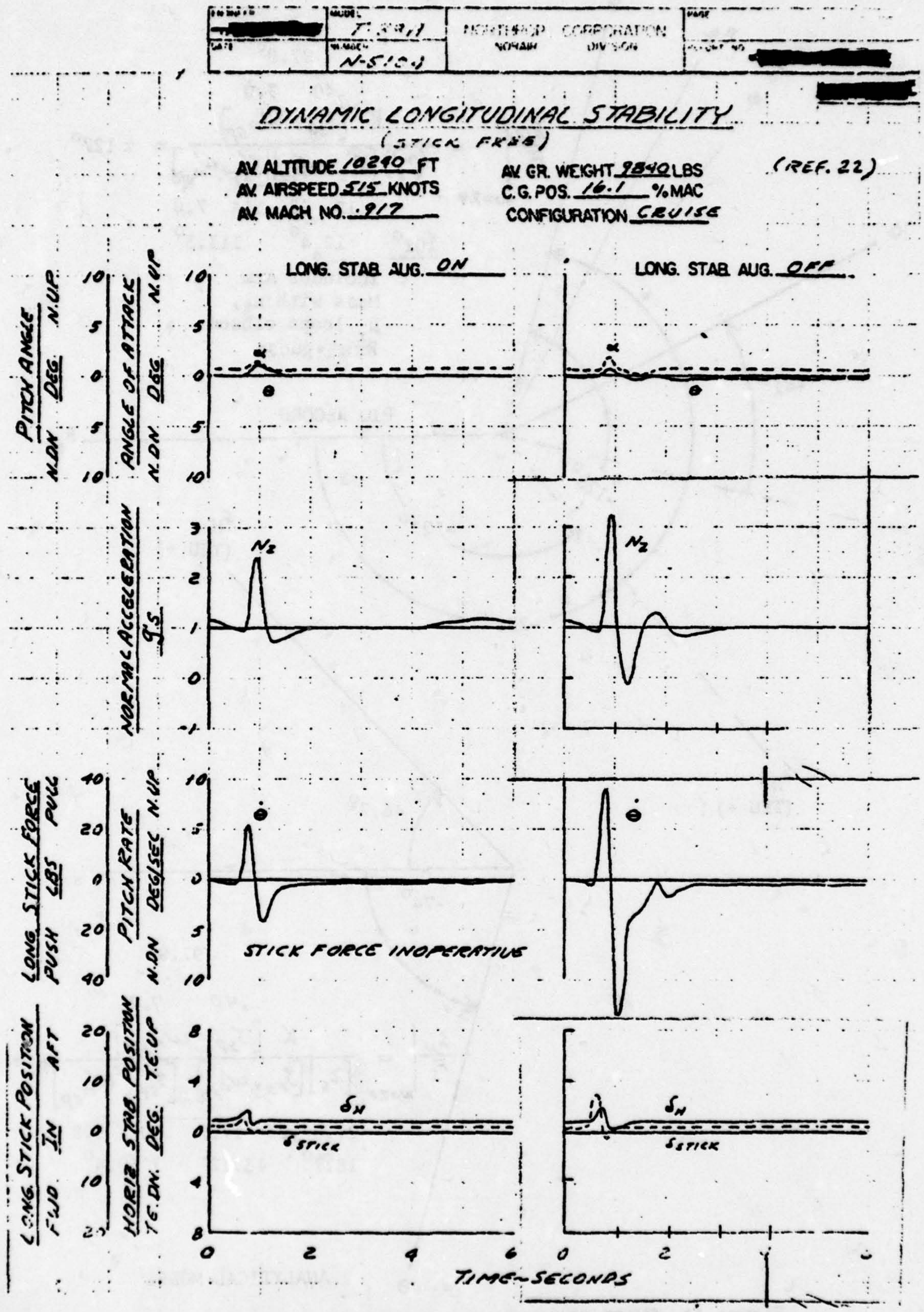


Figure 12 DYNAMIC STABILITY TEST ~ STICK FREE

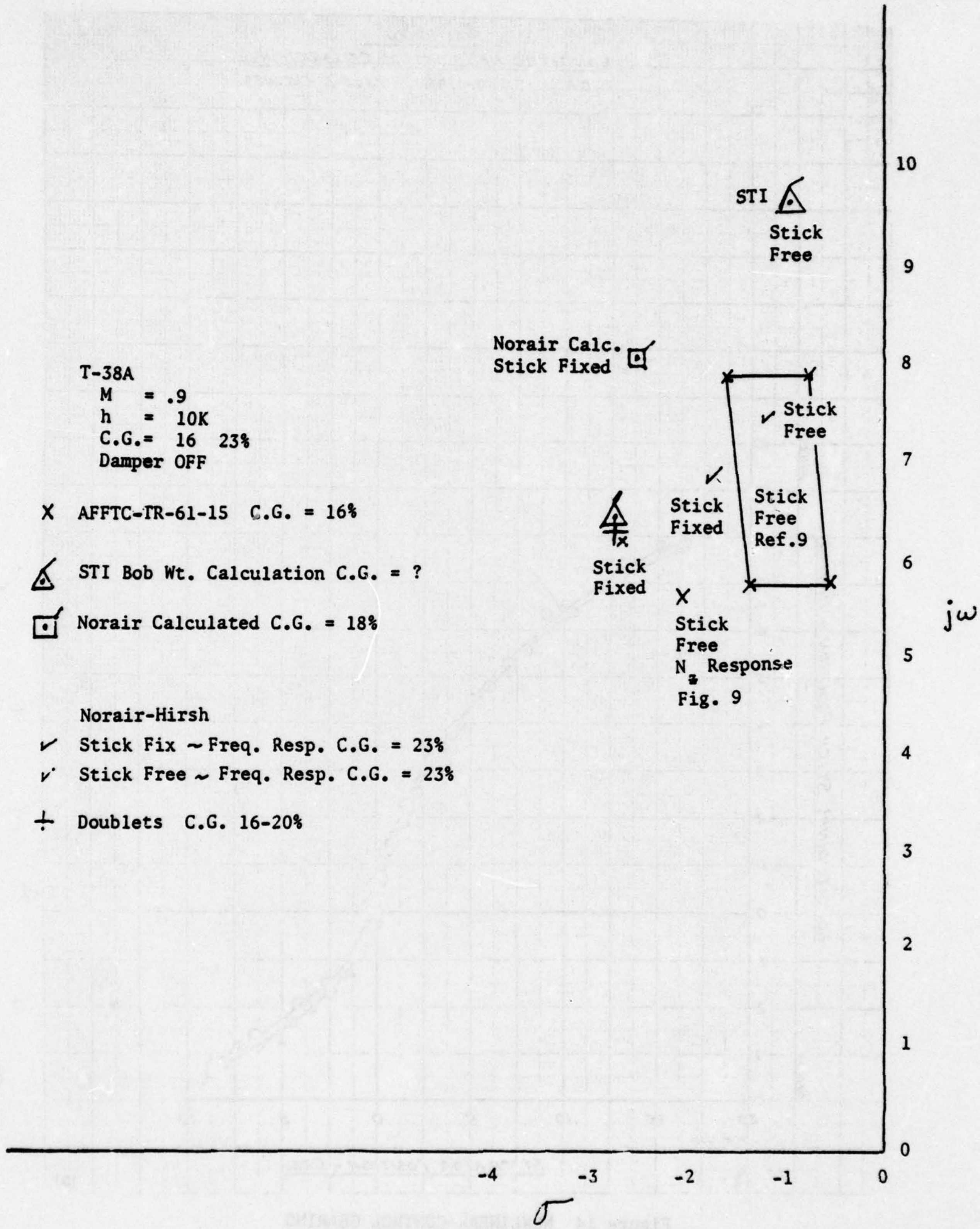


Figure 13 T-38A SHORT PERIOD DYNAMICS

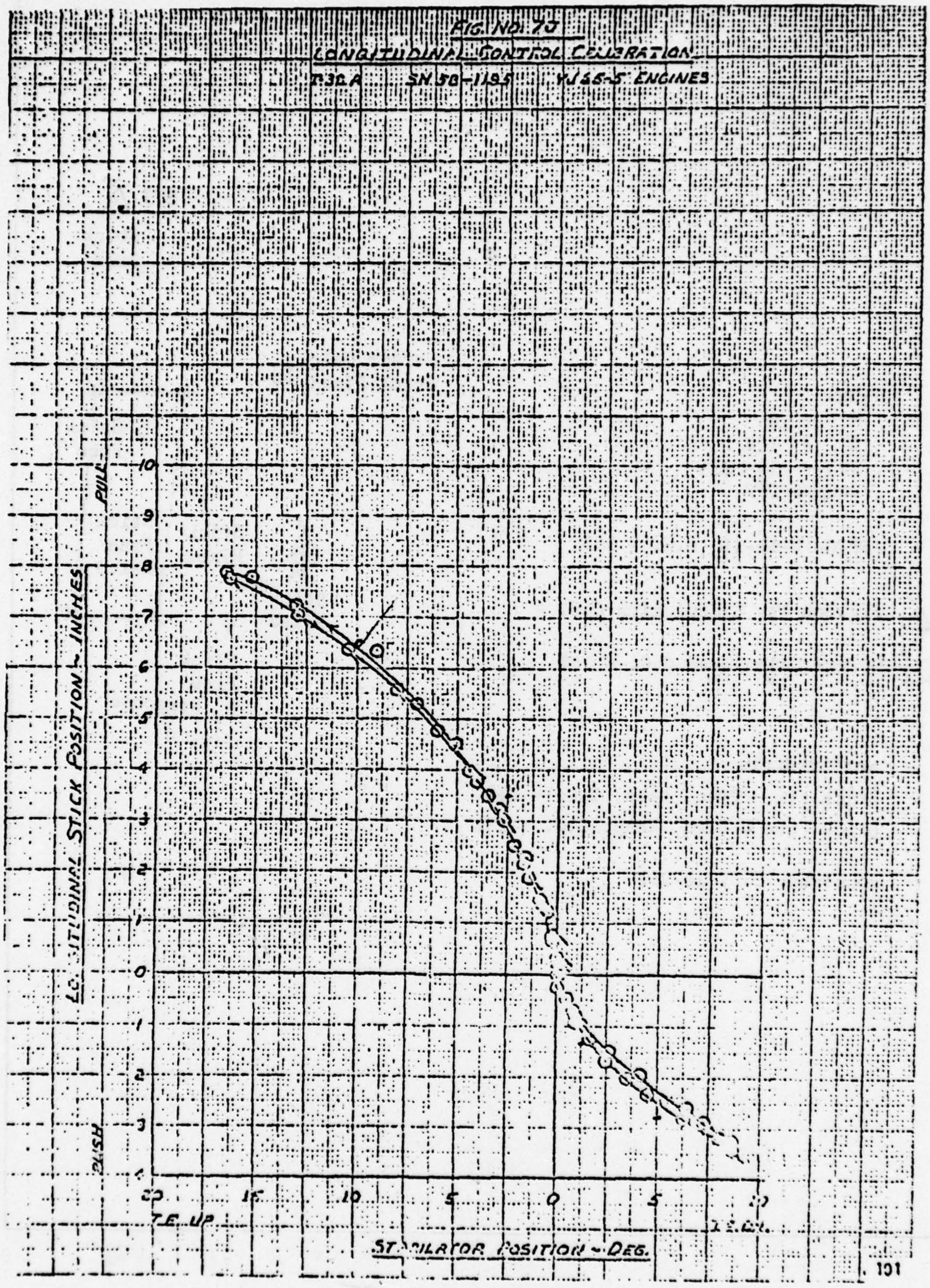


Figure 14 NONLINEAR CONTROL GEARING

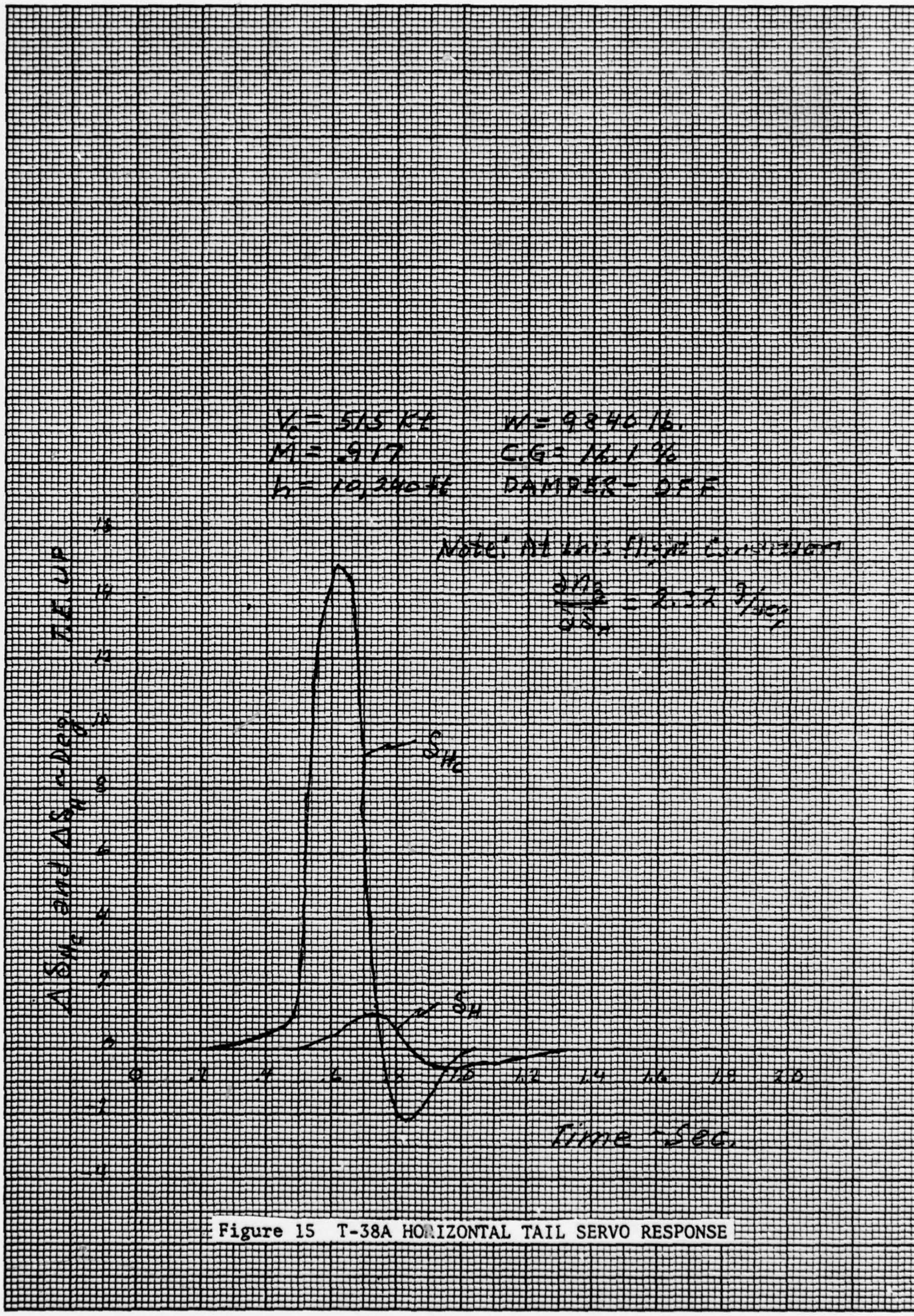


Figure 15 T-38A HORIZONTAL TAIL SERVO RESPONSE

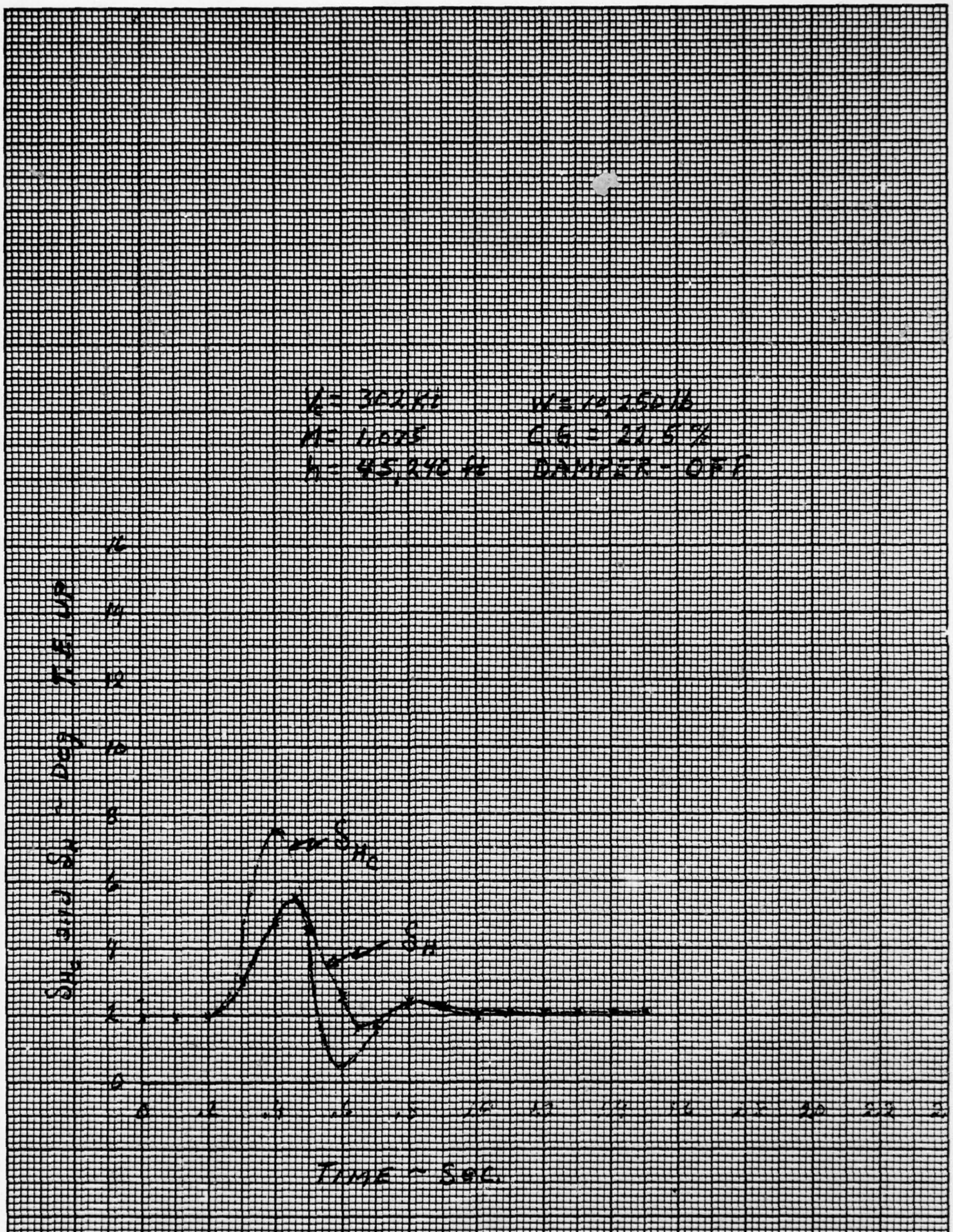


Figure 15a T-38A HORIZONTAL TAIL SERVO RESPONSE

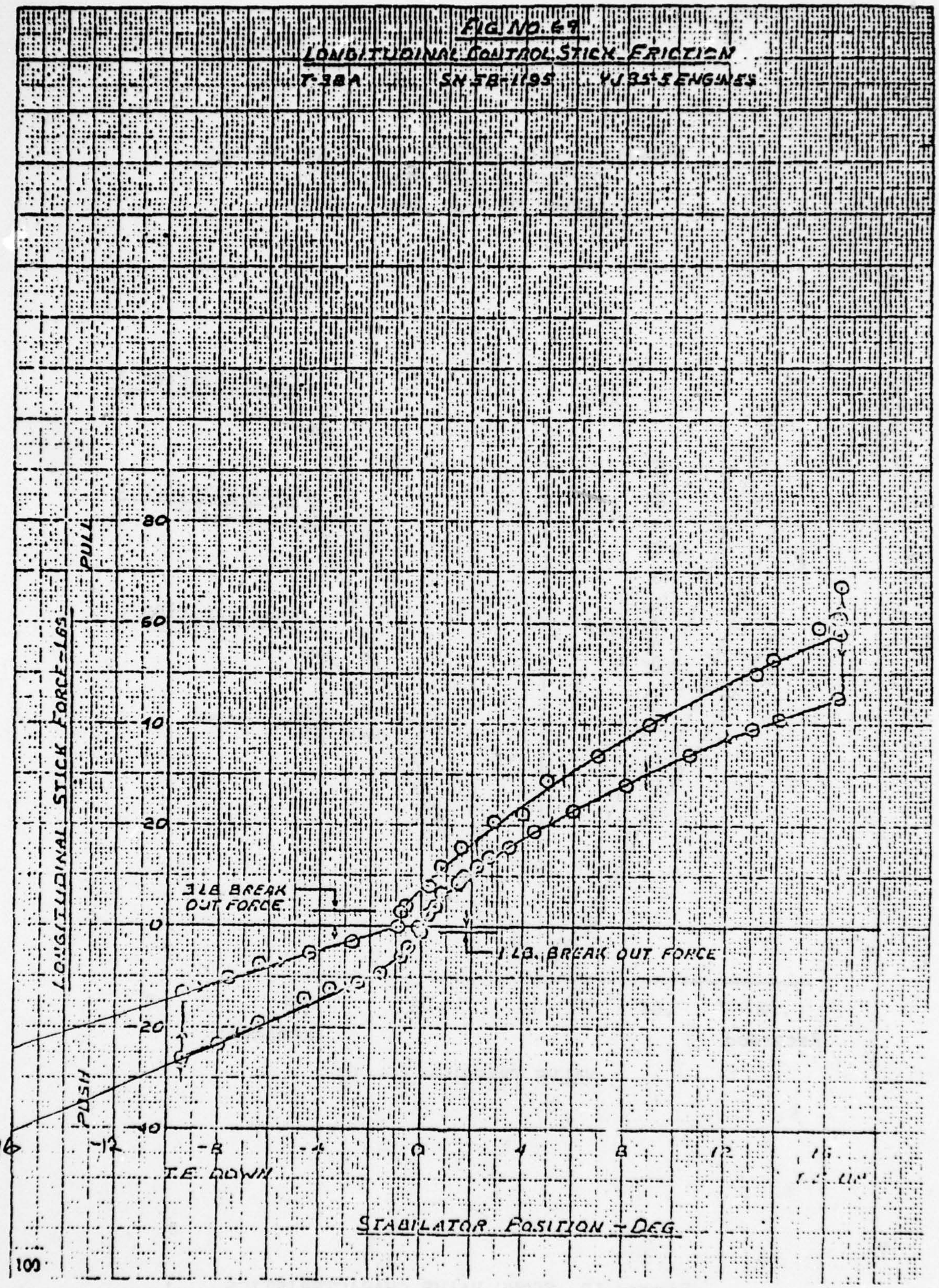


Figure 16 STICK FORCE (APPLIED TO FEEL SPRING)
VS. STABILATOR POSITION

HYDRAULIC FLOW RATE OF THE LONGITUDINAL
CONTROL SYSTEM SERVO VALVES
(Based on Test Bench Data)

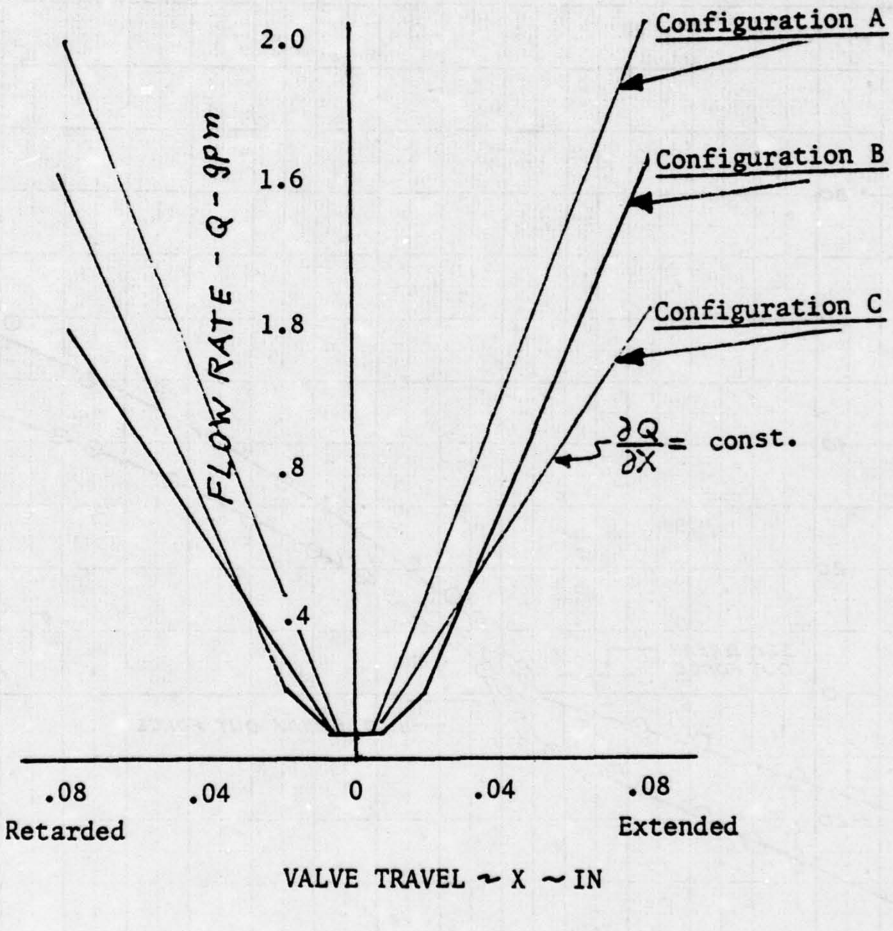


Figure 17 SERVO VALVE CHARACTERISTICS

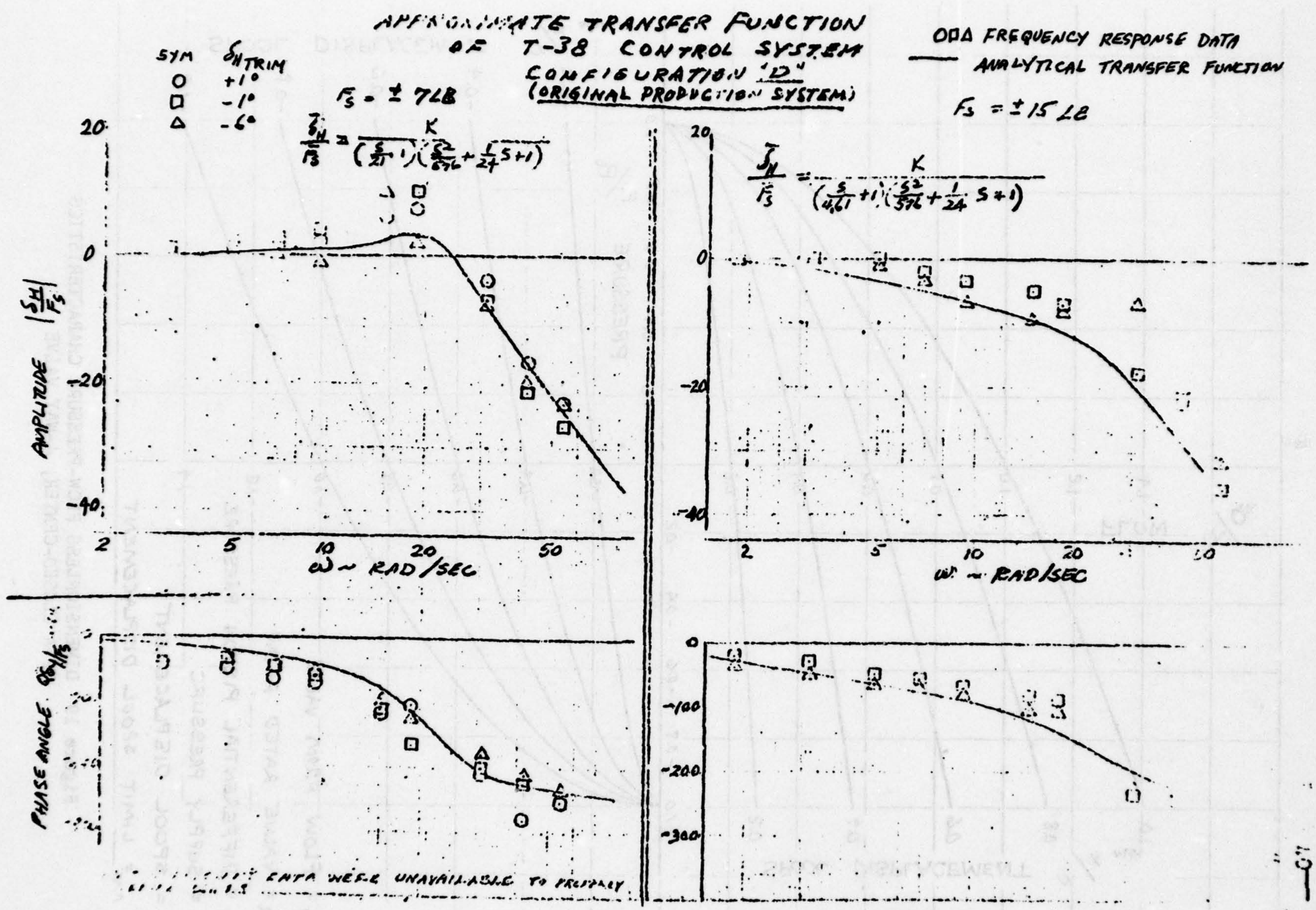


Figure 18 CONTROL SYSTEM FREQUENCY RESPONSE ~ TEST STAND

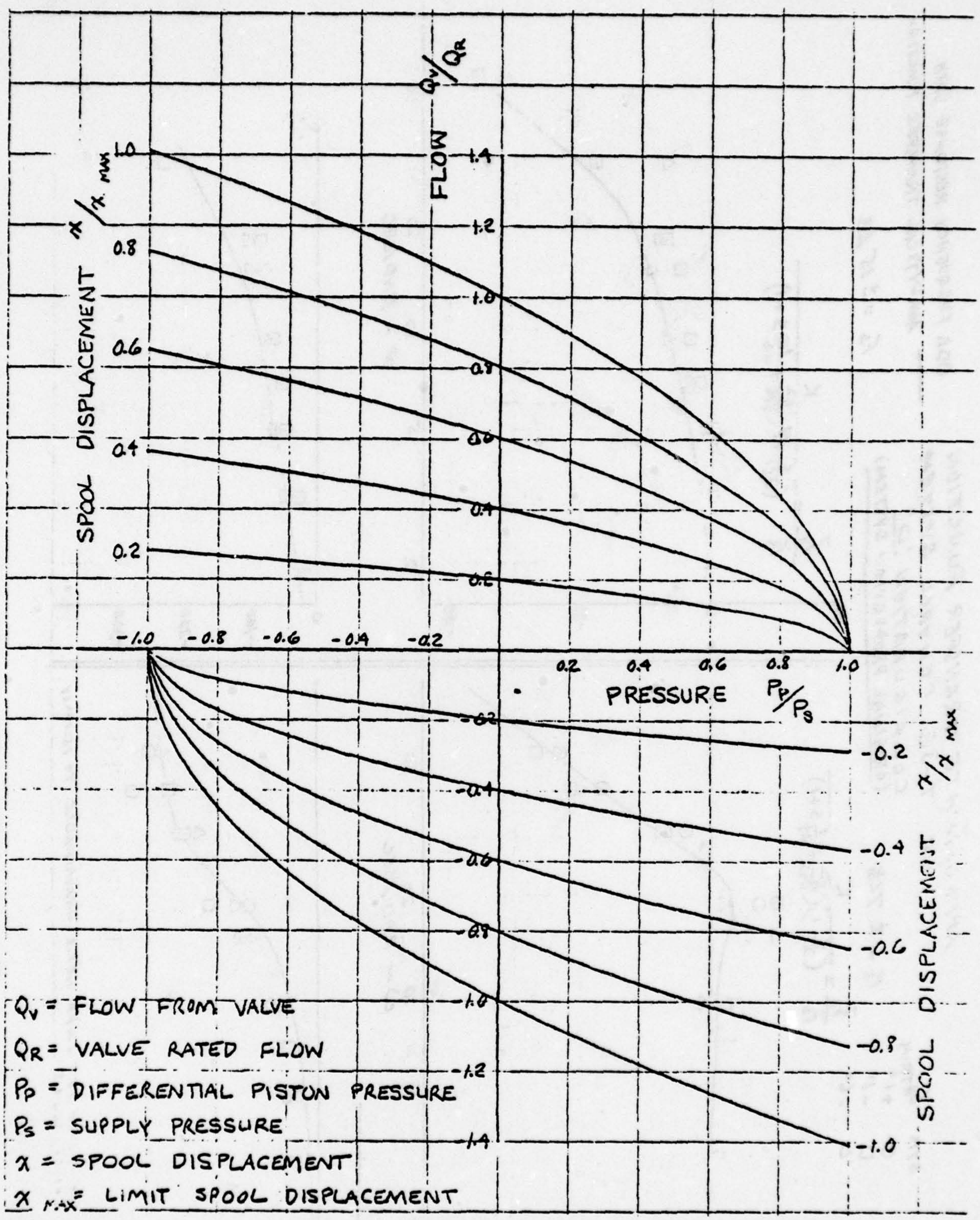


Figure 19 DIMENSIONLESS FLOW-PRESSURE CHARACTERISTICS OF A CLOSED-CENTER, 4-WAY VALVE

Ralph Smith, SRL: Smith pointed out that the phase angle between two time signals can be measured as the distance from the positive peak of one to the next positive peak of the other. He noted further that Chalk had determined the phase between F_s and θ by measuring from the peak of F_s in the pull direction to the peak of θ in the nose-up direction and obtained 255°. Chalk's claim is that this value differs by nearly 180° from that obtained by direct estimation from the STI transfer function. Smith pointed out that Chalk's measurement and subsequent analyses are in error and that the STI results are correct. The error, he claimed, is due to the static gain convention used by STI in the transfer function. Specifically, a push stick force F_s is positive, not a pull force. The fact that a pull stick force yields a nose-up attitude change, is true but irrelevant since the transfer function is predicated on the convention that push F_s is the positive sense. This change of sign accounts for what Chalk claimed is a 180° phase error. It should be noted for the record that the θ/F_s transfer function used by Chalk (p. 14 of his paper) has the wrong sign of the static gain. Since this is proportional to $M_{\delta e}$, and since $M_{\delta e}$ is less than zero in the convention used by STI, then a minus sign should precede the static gain of 0.0084. This same error appears in Smith PIO report (FDL-TR-77-57) where it was, in fact, a typographical omission. When the θ to F_s phase is measured from the T-38 PIO time history as the angle between the peak of F_s in the pull direction and the maximum nose-down θ, then there is no inconsistency between the time traces and the STI time traces. This, Smith claims, is the proper measurement technique.

Chick Chalk, Calspan: [Mr. Chalk expressed disagreement.] I took great care in interpreting sign conventions, trace recording senses on the PIO record and signs of numbers in transfer functions. In drawing the phase diagrams in Figure 11, I chose to relate "pull" force, "nose up" attitude and "trailing edge up" stabilizer. The "pull" force peak is used as the reference. The plus sign on the low frequency gain constant in the θ/F_s transfer function in Table 1 (t.0084) did not result from a typographical error or accident.

I chose to define "pull" force and "nose-up" pitch as positive which is consistent with the convention I used in the phase diagram. In calculating the phase between stabilizer and pitch attitude, θ/δ_H, I defined δ_H "trailing edge up" as positive. Thus pitch attitude "lags" trailing edge up stabilizer by 128° at ω=7.4 rad/sec. In evaluating the transfer function for stabilizer to stick force that is indicated on Figure 11 of my paper, I again used "pull" force and "trailing edge up" stabilizer as references. Thus at low frequency the stabilizer moves "in phase" with the stick force. At ω=7.4 rad/sec the stabilizer will "lead" stick force by 46.7° according to the analytical model but it "lags" stick force by 127° according to the empirical model at the top of Fig 11.

I hope this discussion clarified the situation.

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HIGH ANGLE OF ATTACK FLYING QUALITIES AND DEPARTURE CRITERIA DEVELOPMENT

AN INFORMAL REPORT BY

GARY K. HELLMANN AND

ROBERT B. CROMBIE, 1LT, USAF

CONTROL DYNAMICS BRANCH
FLIGHT CONTROL DIVISION

ABSTRACT

The need for flying qualities and criteria for advanced aircraft is highlighted by the results of a recent survey, and the evolution of USAF high-angle-of-attack flying qualities requirements is documented.

The objectives and some preliminary results of three current AFFDL contracted efforts are presented. Inhouse efforts are also summarized. The goal of the AFFDL high-angle-of-attack program is to correlate desired aircraft behavior with intelligent flying qualities requirements and in turn relate these to suitable design methods. Results will be incorporated into the MIL-PRIME-Standard and Handbook.

Presented at the Symposium and Workshop on Flying Qualities and MIL-F-8785B, Wright State University, Dayton, Ohio 12-15 September 1978.

HIGH-ANGLE-OF-ATTACK FLYING QUALITIES AND
DEPARTURE CRITERIA DEVELOPMENT

by

Gary K. Hellmann
and
Lt Robert B. Crombie

AFFDL/FGC
Control Dynamics Branch
Flight Control Division
Air Force Flight Dynamics Laboratory

Flight in the high-angle-of-attack regime is as common today as it was during man's first attempts at flight. All aircraft experience some degree of high-angle-of-attack flight during the take-off and landing phases while modern fighter aircraft experience even higher angles of attack in air combat maneuvering. During these types of maneuvers, the nonlinear high-angle-of-attack aerodynamic characteristics are likely to be the dominant factors in terms of maneuverability and safety of flight. It is important, therefore, to have a definitive set of flying qualities requirements for this flight regime and the capability to design aircraft which meet or exceed these requirements.

The need for a definitive set of requirements was apparent from the results of a survey (Ref. 1) on the future of research on flying qualities and criteria for highly augmented aircraft. Some important findings of this survey are summarized below:

a. Out of the 41 respondents, 90% indicated that there is a need for more research to better define flying qualities for highly augmented aircraft.

b. Of the above 90%, 92% of the response indicated that for highly maneuverable aircraft, this need was either important or critical, with 32% mentioning a need for more research in high-angle-of-attack and stall/spin characteristics specifically.

c. When asked to rate the importance of several areas of flying qualities research, 73% rated research on the limit-of-performance regions (such as high-angle-of-attack maneuvering) as high with 54% rating research on nonlinear flight regimes (such as large-amplitude maneuverability) as high.

With such a high percentage of the responses indicating the need for

more research in high-angle-of-attack flying qualities, one must wonder how this regime is treated in the current flying qualities specification, MIL-F-8785B(ASG), "Flying Qualities of Piloted Aircraft" (Ref. 2).

One of the results of the December 1971 Stall/Post-Stall/Spin Symposium (Ref. 3) was the change in emphasis from requirements for developed spins and their recovery to the need for departure resistance. This redirection was reflected in the March 31, 1971 Amendment 1 to MIL-8785B. This amendment was largely qualitative in its characterization of the desirable high-angle-of-attack flying qualities. It emphasized resistance to departure and simple recovery along with loss-of-control warning. Some of these qualitative requirements were also accompanied by quantitative limits on such variables as sideslip, oscillatory bank angle, stall onset rate, etc. Most of these quantitative aspects were also incorporated in MIL-S-83691 (USAF), 31 March 1971 (Table I) and MIL-S-83691A (USAF), 15 April 1972, "Stall/Post-Stall/Spin Flight Test Demonstration Requirements for Airplanes," (Ref. 4). When Amendment 2 to MIL-F-8785B, 15 April 1973, was prepared, the quantitative requirements were deleted and its stall/post-stall/spin requirements remained primarily qualitative in nature. Amendment 2 does contain the current high-angle-of-attack flying qualities requirements (Ref. 5).

Since the evolution of the above amendments to the present specification, there have been major developments in flight control system design including new control devices, aerodynamic designs, air combat tactics, and an enhanced appreciation for the desirability of high-angle-of-attack flight. The current specification does not reflect any of these major developments. The proposed Amendment 3 to MIL-F-8785B, while still qualitative in nature, does begin to address some of the aforementioned developments. However, with the approach of the "MIL-PRIME-STD" and the supporting Handbook ("MIL-HDBK"), significantly more quantitative research is required in order to satisfactorily determine desirable high-angle-of-attack flying qualities requirements and appropriate design criteria. The AFFDL has a series of ongoing and planned research and development efforts that will attempt to provide some of the needed quantitative data. Each of these efforts is summarized below.

IDENTIFICATION OF KEY MANEUVER-LIMITING FACTORS, AFFDL CONTRACT NO. F33615-76-C-3072, SYSTEMS TECHNOLOGY, INC.

OBJECTIVE/APPROACH:

The objectives of this effort are to identify key design parameters that limit high-angle-of-attack maneuverability for several high-performance fighter aircraft and to postulate fundamental aerodynamic and control system design methodologies that will alleviate the limiting conditions. The key parameters and the design methodology will be validated and further investigated in a manned simulation of the aircraft and control

system dynamics used in the analysis. Methods and criteria will be developed for evaluation of handling qualities describing high-angle-of-attack maneuvering flight. Consideration will be given to the following factors:

a. Applicability of the Cooper-Harper rating scale to the pilot tasks associated with high-angle-of-attack maneuvering flight in and about the limiting conditions. Potential definition of a scale more suited to description of handling qualities near the limiting conditions.

b. Fixed-base vs. moving-base simulation. Displays and instrumentation required for evaluation of handling qualities associated with the high-angle-of-attack maneuvering flight.

c. Definition of simulation tasks required to best evaluate handling qualities associated with the high-angle-of-attack maneuver-limiting conditions.

The results of this study will be used to formulate generalized design guides, handling qualities criteria and requirements for high-angle-of-attack maneuvering flight which can be incorporated into MIL-F-8785B.

PROGRESS:

This contract was awarded to Systems Technology, Inc. (STI), in May 1976, to investigate the F-4 and F-14 aircraft. The limiting conditions for each aircraft and associated causal factors were identified, but have not been validated through a manned simulation. For the F-4, the limiting factors and associated causal factors are:

wing rock

- unstable Dutch roll

- nonlinear aerodynamic interaction

yaw SAS ineffective

$\omega_r > \omega_d$

$\omega_{SR} < 1/T_{\theta_3}$

- improve via

$a_y \rightarrow \delta_r$

$p \rightarrow \delta_a$

roll control via lateral stick

- divergent because ω_ϕ^2 negative

- correct with stick-rudder crossfeed

nose slice divergence

- nonminimum-phase zero, $N^{\theta}_{\delta_e}$

- due to N'_{α}, L'_{α}

- open loop or push stick to recover.

For the F-14, the limiting factors and associated causal factors are:

wing rock

- unstable Dutch roll (small C_{ℓ_p}, negative C_{n_p}, small negative C_{n_β})

roll reversal

- negative $C_{n_{\delta a}}(\omega_{\phi}^2)$

key flight control consideration (preliminary analysis)

- augment C_{ℓ_p} via roll rate CAS

- augment C_{n_β}, C_{n_p} via sideslip stability augmentor

- reduce $C_{n_{\delta a}}$ via aileron to rudder interconnect.

Although STI has not performed the manned simulation, they have developed the definition of simulator tasks, a tentative departure rating scale, and a questionnaire for the overall assessment of the configuration. The simulation tasks are presented in Table II. The pilot rating scale is shown in Figure 1 with the assessment questionnaire presented in Table III.

Overall program results will be available in August 1979.

HIGH-ANGLE-OF-ATTACK DESIGN GUIDES AND FLYING QUALITIES CRITERIA, AFFDL CONTRACT NO. F33615-78-C-3604, SYSTEMS TECHNOLOGY, INC.

OBJECTIVE/APPROACH:

This effort consists of both gathering and analyzing existing data required to provide new and/or improved high-angle-of-attack flying qualities requirements and design guides. An extensive literature survey and a comprehensive series of personal visits to various organizations in the aeronautical community will be used to revise high-angle-of-attack

flying qualities specifications. The design guides effort consists of gathering high-angle-of-attack design guides from several sources and performing appropriate analysis for verification. This effort will include both aerodynamic and control system design methodologies. Aerodynamic data and flight characteristics will be used for correlation with existing or potential new guides. Results will be presented in compendium form. Future areas for technical development will be suggested and justified. Consideration will be given to the relationship of application of guides to obtaining adequate flying qualities.

PROGRESS:

This contract was awarded to Systems Technology, Inc. in April 1978. The extensive literature search and survey of designers and users are currently being conducted.

THE DEVELOPMENT OF AN AIRCRAFT DEPARTURE CRITERION AND ASSOCIATED DESIGN CHARTS, CONTRACT NO. N62269-77-C-0106/DEPARTURE TRENDS FOR CCV AIRCRAFT, CONTRACT NO. F33615-78-C-3600, BIHRLE APPLIED RESEARCH, INC.

OBJECTIVE/APPROACH:

The objective of the NADC effort was to generate design charts and associated boundaries for identifying departure and uncoordinated roll-reversal flight characteristics as a function of three aerodynamic parameters: C_{ℓ_β}, C_{n_β}, and $C_{n_{\delta_a}}$, including several variations in each.

The objective of the AFFDL effort is to investigate departure characteristics of an aircraft that is statically unstable in pitch and to determine if significant trends can be correlated with aerodynamic characteristics and variations in the flight control system. Three levels of static instability will be studied using the aerodynamic characteristics developed for the NADC contract as a basis for this investigation. Since the basic configuration did not require a flight control system, the contractor is required to design a simple longitudinal system to artificially provide the original level of static stability. The resulting control system will also include a simple angle-of-attack limiting system. The effects of the C_{ℓ_β}, C_{n_β}, and $C_{n_{\delta_a}}$ variations developed in the NADC effort and the influence of the developed flight control augmentation system will be determined for the unstable configurations. The results of this effort will be correlated with those of the NADC effort to determine trends in the departure characteristics of statically unstable aircraft.

PROGRESS:

Both contracts were awarded to Bihrle Applied Research, Inc. The NADC effort was completed in July 1978, with a final report submitted at that time. The AFFDL contract was awarded in May 1978.

NADC CONTRACT - FINAL RESULTS (Ref. 6)

A departure boundary for each level of $C_{n_{\delta a}}$ is shown in Figure 2. For combinations of C_{n_β} and C_{ℓ_β} above the boundary, departure does not occur. For combinations below the boundary, departure occurs for the maneuvers used in this study. Using the bank angle information obtained in the development of the design charts, roll-reversal boundaries were developed. These boundaries, shown in Figure 3, are not departure boundaries but uncoordinated roll-reversal boundaries requiring the pilot to apply rudder to coordinate the maneuver.

AFFDL CONTRACT

Since the contract was begun in May, limited progress can be reported at this time. The effects of variations in C_{m_q} and C_{m_β} were investigated in order to determine their influence on the departure boundaries developed under the NADC effort. Preliminary results indicate the following:

a. Reducing the C_{m_q} level could result in a significant alteration of the departure boundaries.

b. A positive C_{m_β} level could alter the departure boundary slightly.

c. A negative C_{m_β} would more seriously change the boundary.

AERODYNAMIC HYSTERESIS EFFECT ON FLYING QUALITIES - INHOUSE EFFORT

OBJECTIVE/APPROACH:

One of the second-order (nonlinear) effects that may significantly change an aircraft's flying qualities at high angles of attack is aerodynamic hysteresis. By making the aerodynamic coefficients double-valued functions, changes occur in derivative definition and zero β data (trim requirements, control power). The objective of this effort has been to determine the effects of an idealized hysteresis loop and to define hysteresis uncovered in wind tunnel data (AFFDL-TM's 76-75-FGC and 78-91-FGC). Future efforts will involve further wind tunnel studies and mathematical definition of the phenomenon and its effects on pilot-in-the-loop simulations.

PROGRESS:

Hysteresis has been shown to be a potential cause of wing rock (TM-76-74) and some examples of its existence in wind tunnel data have been found (TM-78-91). Further work may be appended to wind tunnel tests or simulations as the opportunities arise. Work in the area by other organizations will be encouraged and supported if resources are available.

ROLL/PITCH INERTIAL COUPLING OF A STATICALLY UNSTABLE AIRCRAFT - INHOUSE EFFORT

OBJECTIVE/APPROACH:

The objective of this effort is to determine the effect of various levels of stability-axis yaw rate (αp-r) feedback on the roll/pitch coupling departures of a statically unstable lightweight-fighter configuration at low airspeeds and high roll rates. The influence that these levels have on other high-angle-of-attack maneuvers will also be assessed. Both digital computer and pilot-in-the-loop simulations will be used for the analysis. By varying the gain K on the K(αp-r) feedback loop, a check will be made to determine any change in the aircraft's roll characteristics and stabilator movement. The maximum stabilator displacement limit and the center of gravity location will be varied in order to identify their effects on the inertial coupling during the rolling maneuvers. The manned simulation task will provide additional evaluation through the investigation of several maneuvers. The simulation pilots will also rate the handling qualities of the aircraft for each maneuver using both the Cooper-Harper and several potential high-angle-of-attack rating scales.

PROGRESS:

A digital computer simulation at the AFFDL and a manned simulation at the AFFTC have been completed. Results from the two simulations have not been correlated at this time. Based on some preliminary comments from the AFFTC, the pilots have reacted favorably to the new rating scales. More detailed comments from the pilots will be forwarded to Systems Technology, Inc. to be included in their work on deriving a new high-angle-of-attack rating scale.

EVALUATION OF CRITERIA TO PREDICT LATERAL-DIRECTIONAL DIVERGENCE AT HIGH-ANGLES-OF-ATTACK - INHOUSE EFFORT

OBJECTIVE/APPROACH:

The $C_{n_{\beta DYN}}$, aileron-alone and lateral-control departure parameters

will be calculated for several levels of lateral-directional static stability characteristics. A comparison will then be made in an attempt to correlate these parameters with computer-generated departure susceptibility time histories of a statically unstable lightweight-fighter configuration incorporating an advanced flight control system. The calculated departure parameters will also be compared to the migration of the roots of the lateral-directional characteristic equation. Based upon the results of this investigation, conclusions on the use of the departure parameters in the presence of a flight control system and recommendations for future work will also be made.

PROGRESS:

Two $C_{n\beta}$, $C_{\ell\beta}$ combinations were defined that would result in significantly different lateral-directional stability characteristics. These combinations were used to develop the $C_{n\beta_{DYN}}$ (Fig. 4), aileron-alone (Fig. 5) and lateral-control departure parameters. These parameters by themselves and with the crossplot of $C_{n\beta_{DYN}}$/aileron-alone departure parameter (Fig. 6) and $C_{n\beta_{DYN}}$/lateral-control departure parameter were used to predict where lateral-directional departures will occur.

In order to assess the ability of these parameters to predict departures for this configuration, a digital computer simulation was conducted using the two $C_{n\beta}$, $C_{\ell\beta}$ combinations. The simulation was performed with the longitudinal and lateral-directional control system feedbacks open and also with the lateral-directional feedbacks closed. These two levels of augmentation were investigated in order to determine if significant changes in the departure characteristics would be obtained. The simulation task involved a simple 360° roll about trim points at high angles of attack. The resulting motions were analyzed and correlated with the predicted departure characteristics. These same motions were also correlated with the root migration of the lateral-directional characteristic equation (Figs. 7, 8).

Final analysis and correlation of this effort has not been completed at this time. Results, when available, will be published in AFFDL-TM-78-56-FGC.

DEPARTURE CRITERIA VALIDATION AND EVALUATION - PROPOSED INHOUSE EFFORT

OBJECTIVE/APPROACH:

Criteria relating departure from controlled flight to various stability and control parameters have been developed (Refs. 6, 7) using certain "typical" configurations and for varying degrees of linearization. In order for the aircraft designer to confidently use these criteria, they must be validated for different flight maneuvers and stability and

control characteristics. In addition, the designer needs to know which parameters are most crucial for departure resistance or recovery. He also needs to know how accurately they need to be calculated during the design phase. Existing computation methods (e.g., Datcom, FLEXSTAB) are currently not appropriate for use in the high-angle-of-attack regime. An evaluation will therefore be made to determine the most crucial design parameters in terms of a configuration's departure sensitivity as well as how accurately they must be determined by revised computation methods in terms of variations with angle of attack and sideslip.

The models, maneuvers, and assumptions used in determining the various departure criteria will be compared and contrasted with other available knowledge. Where appropriate, further criteria development or validation will be made via sensitivity analyses on a six-degree-of-freedom computer program. Aircraft mathematical models will also be devised to test for the most crucial designer-estimated parameters. These models will include non-linearities with angle of attack and sideslip. The sensitivity of the resulting aircraft motion and departure susceptibility to variations in the estimated parameters will be determined.

BENEFITS:

Besides providing guidance to aircraft designers on what makes a departure-resistant configuration, this effort will also contribute to two other elements of the Flight Dynamics Laboratory's stability and control research. It will identify the most crucial parameters which need to be predicted with a quantifiable accuracy by improved design predictions techniques. In addition, an understanding of discrepancies between stability and control parameters derived from prediction methods, wind-tunnel measurements, and flight tests will be developed for the high-angle-of-attack regime in conjunction with a broad, new stability and control data discrepancies effort.

CONCLUSION:

In summary, the Control Dynamics Branch of the Flight Dynamics Laboratory is undertaking to identify and solve some of the major problems that degrade the flying qualities of military aircraft in the high-angle-of-attack regime. Closed-loop linearized work that applies to task-oriented flying qualities is being done by contractors such as STI. Open-loop nonlinear analysis pertaining to departure susceptibility is being done both in-house and on contract. Problems involving statically unstable aircraft are also being addressed. Work sponsored by other government agencies will be monitored through the DOD/NASA Stall/Spin Coordinating Committee.

Over the next few years, we hope to develop a good understanding of

the state of the art in high-angle-of-attack flying qualities, airframe and flight control system design, and departure resistance criteria. This will enable us to correlate the desired aircraft behavior at high angles of attack with intelligent flying qualities specifications. The end product will be much needed quantitative requirements and guidance to incorporate into the coming Mil-Prime-Standard and its Handbook.

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TABLE I

FLIGHT TEST DEMONSTRATION MANEUVERS

Test Phase	Maneuver Requirements					
	Control Application	Entry Conditions[1]				
		Smooth AOA Rate[4]		Abrupt AOA Rate[5]		Tactical[6]
		One g	Accelerated[3]	One g	Accelerated[3]	
A Stalls	Pitch control applied to achieve the specified AOA rate, lateral/directional controls neutral or small lateral/directional control inputs as normally required for the maneuver task. Recovery initiated after the pilot has a positive indication of: (a) A definite g-break, or (b) a rapid angular divergence, or (c) the aft stick stop has been reached and AOA is not increasing.	Class: I, II, III, IV	Class: I, II, III, IV	Class: I, II, III, IV	Class: I, IV	Class: I, IV
B Stalls with Aggravated Control Inputs	Pitch control applied to achieve the specified AOA rate, lateral/directional controls as required for the maneuver task. When condition (a), (b), or (c) from above has been attained, controls briefly misapplied,[7] intentionally or in response to unscheduled aircraft motions, before recovery attempt is initiated.	Class: I, II, III, IV	Class: I, II, III, IV	Class: I, II, III, IV	Class: I, IV	Class: I, IV
C Stalls with Aggravated and Sustained Control Inputs	Pitch control applied to achieve the specified AOA rate, lateral/directional controls as required for the maneuver task. When condition (a), (b), or (c) has been attained, controls are misapplied[7,9], intentionally or in response to unscheduled aircraft motions, and held for 3 seconds[8,9] before recovery attempt is initiated.	Class: I, II, III, IV	Class: I, II, III, IV	Class: I, II, III, IV	Class: I, IV	Class: I, IV
D Spin Attempts[2,10] (this Phase required only for training aircraft which may be intentionally spun and for Class I and IV aircraft in which sufficient departures or spins did not result in Test Phases A, B, or C to define characteristics)	Pitch control applied abruptly, lateral/directional controls as required for the maneuver task. When condition (a), (b), or (c) has been attained, controls applied in the most critical positions to attain the expected spin modes of the aircraft and held for up to 15 seconds before recovery attempt is initiated, unless the pilot definitely recognizes a spin mode.			Class: I, IV	Class: I, IV	Class: I, IV

TABLE II
TASK DEFINITIONS FOR RATING HIGH-ANGLE-OF-ATTACK MANEUVERS

1. Straight Ahead Stall
   a. Keep wings level.
   b. Maintain constant heading.
   c. Continue stick ramp until one of the following occurs:

      (1) Definite g-break occurs
      (2) Aft stick limit is reached
      (3) Aircraft departs

   These events shall constitute the definition of "stall" in the following task descriptions.

2. Constant Attitude Stall
   a. Pitch to and maintain constant value of pitch attitude.
   b. Reduce power until stall occurs.
   c. Maintain constant heading.
   d. Keep wings level.

3. Bank-to-Bank Turns
   a. Establish 60-degree bank turn.
   b. Reverse direction using rapid roll rate to 60 degrees in other direction.
   c. Hold pitch attitude approximately constant.
   d. Repeat maneuver increasing pitch attitude each time until stall occurs.
   e. When stall occurs recover to wings-level flight with heading approximately equal to heading at stall.
   f. Rating should pertain to Task 3e

4. Wind-Up Turns
   a. Initiate moderate roll rate.
   b. Maintain altitude approximately constant by increasing back pressure until stall occurs.
   c. When stall occurs, recover to wings-level flight with heading approximately equal to heading at stall.
   d. Rate Task 4c.

5, 6 Track Target (5→steady climb; 6→bank-to-bank turns)
   a. Employ maximum effort to stay on target's tail; however, do not consider ability to keep pipper on target in ratings.
   b. When aircraft stalls, your task is to recover to wings-level flight in same general direction as when stall occurred in the minimum time.
   c. Angular excursions should start decreasing shortly after recovery controls are applied.
   d. Performance should take into account the peak angular and angular rate excursions at stall.

TABLE III
OVERALL ASSESSMENT OF CONFIGURATION

1. Departure Resistance

   a. Extremely susceptible to departure.

   b. Susceptible to departure.

   c. Resistant to departure.

   d. Extremely resistant to departure.

2. Departure Warning

   Discuss warning characteristics with respect to accomplishment of mission. Some aspects of this are:

   a. Does warning allow me to utilize a greater portion of the flight envelope?

   b. Does warning interfere with the primary task?

   c. Is warning consistent across all types of entry? Is this type of consistency even desirable?

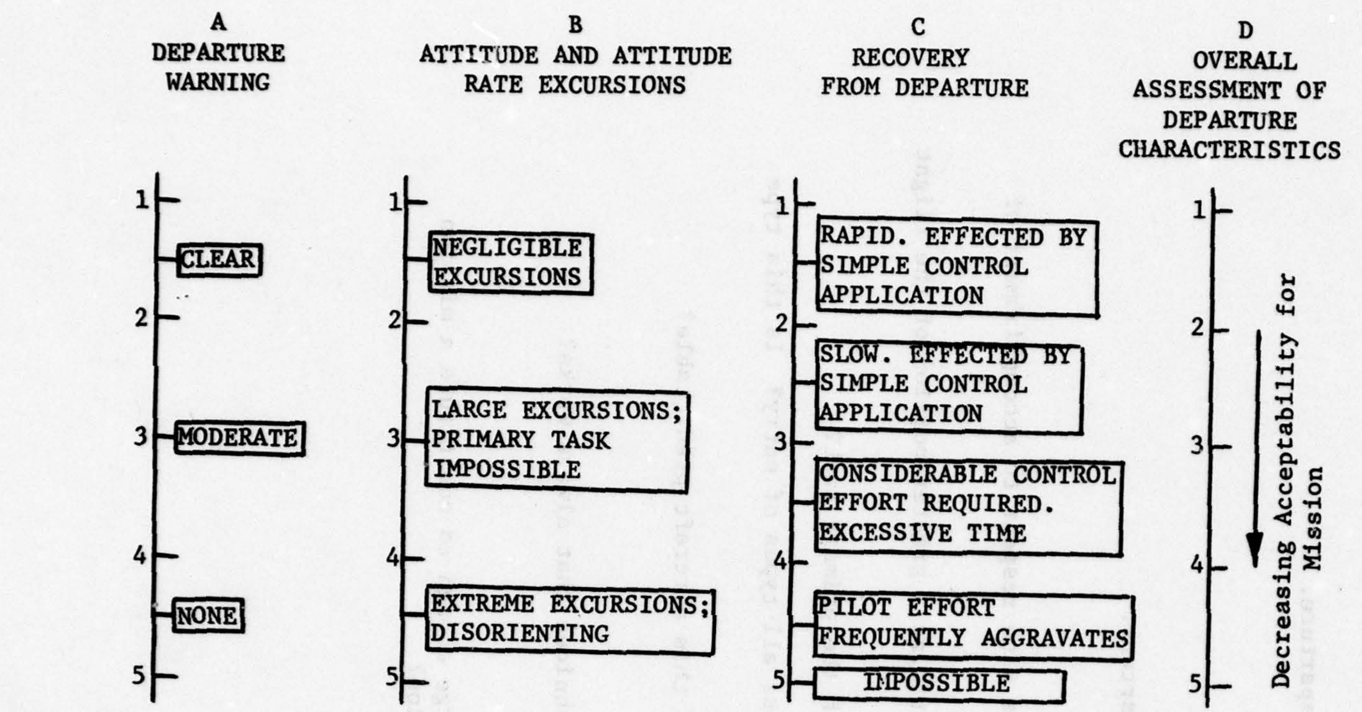
   d. Are the departure modes of this aircraft predictable?

3. Departure Recovery

   Is there a unique recovery technique that always works?

4. Departure Severity

   Are departure motions ever large enough to constitute a mission hazard worthy of a flight restriction?



- Describe warning
- Did warning interfere with primary task? If so, to what extent?

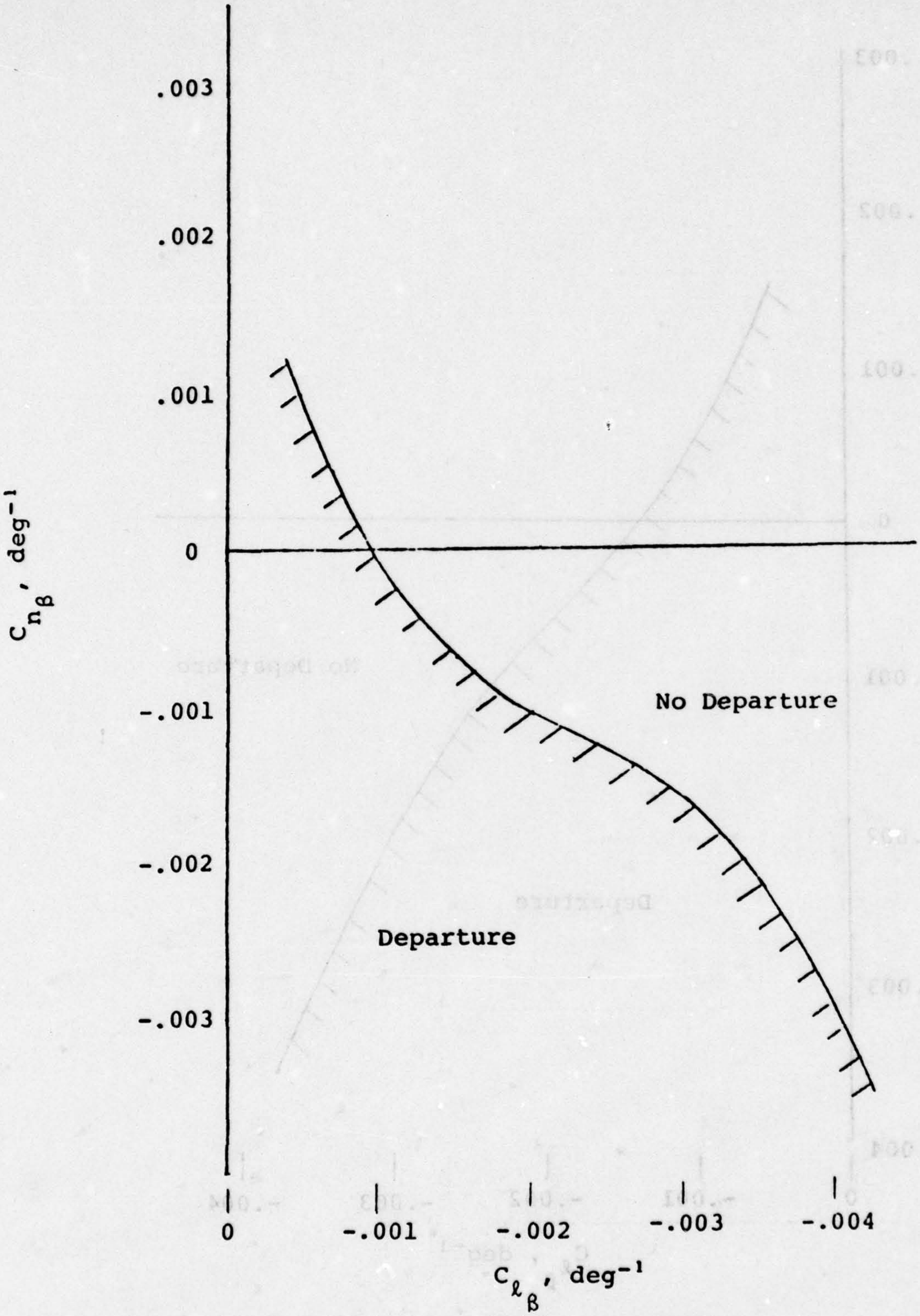
- Describe departure sequence
  1. Nose slice
  2. Wing rock
  3. Rolling departure
  4. Pitch up
- What was primary departure variable? ($\theta, \dot{\theta}, \psi, \dot{\psi}, \phi, \dot{\phi}$)

- Was attitude loss primarily due to a path divergence or to attitude excursions?

- Fighter
- Trainer

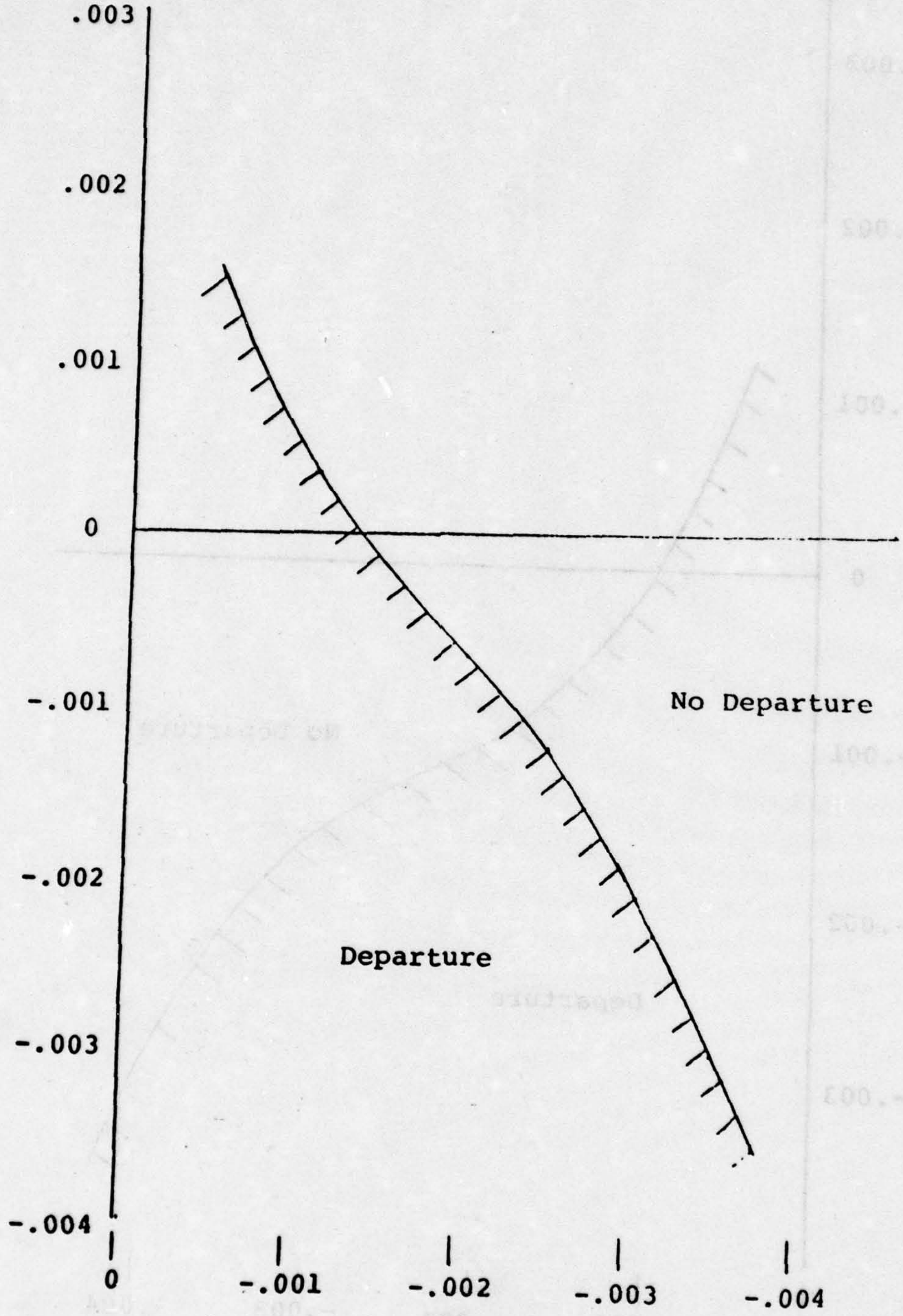
FIGURE 1

TENTATIVE DEPARTURE RATING SCALE



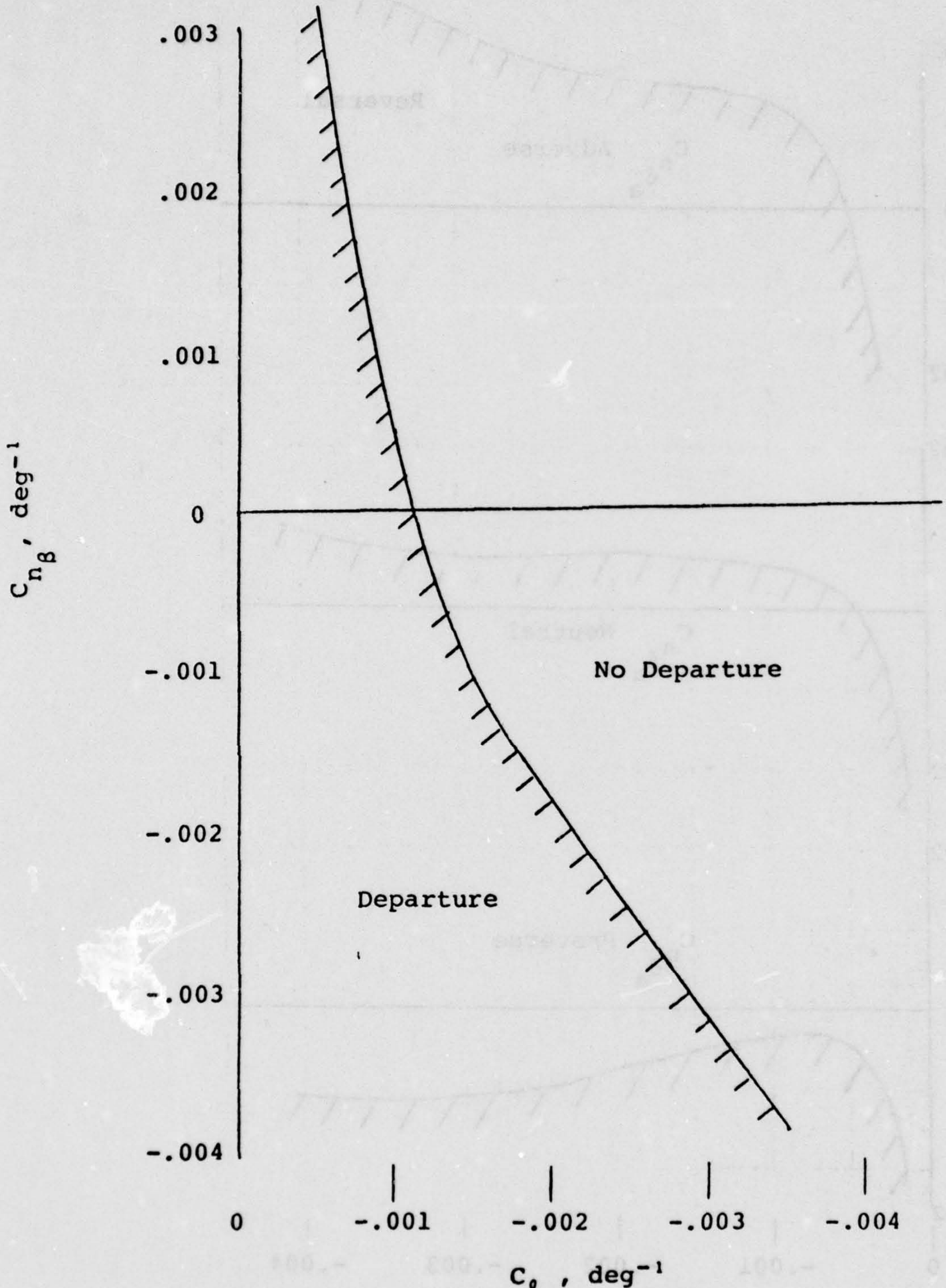
(a) Adverse $C_{n_{\delta_a}}$

Figure 2.- Departure boundaries. (From Ref 6)



(b) Neutral $C_{n_{\delta_a}}$

Figure 2.- Continued.



(c) Proverse $C_{n_{\delta_a}}$

Figure 2.- Concluded.

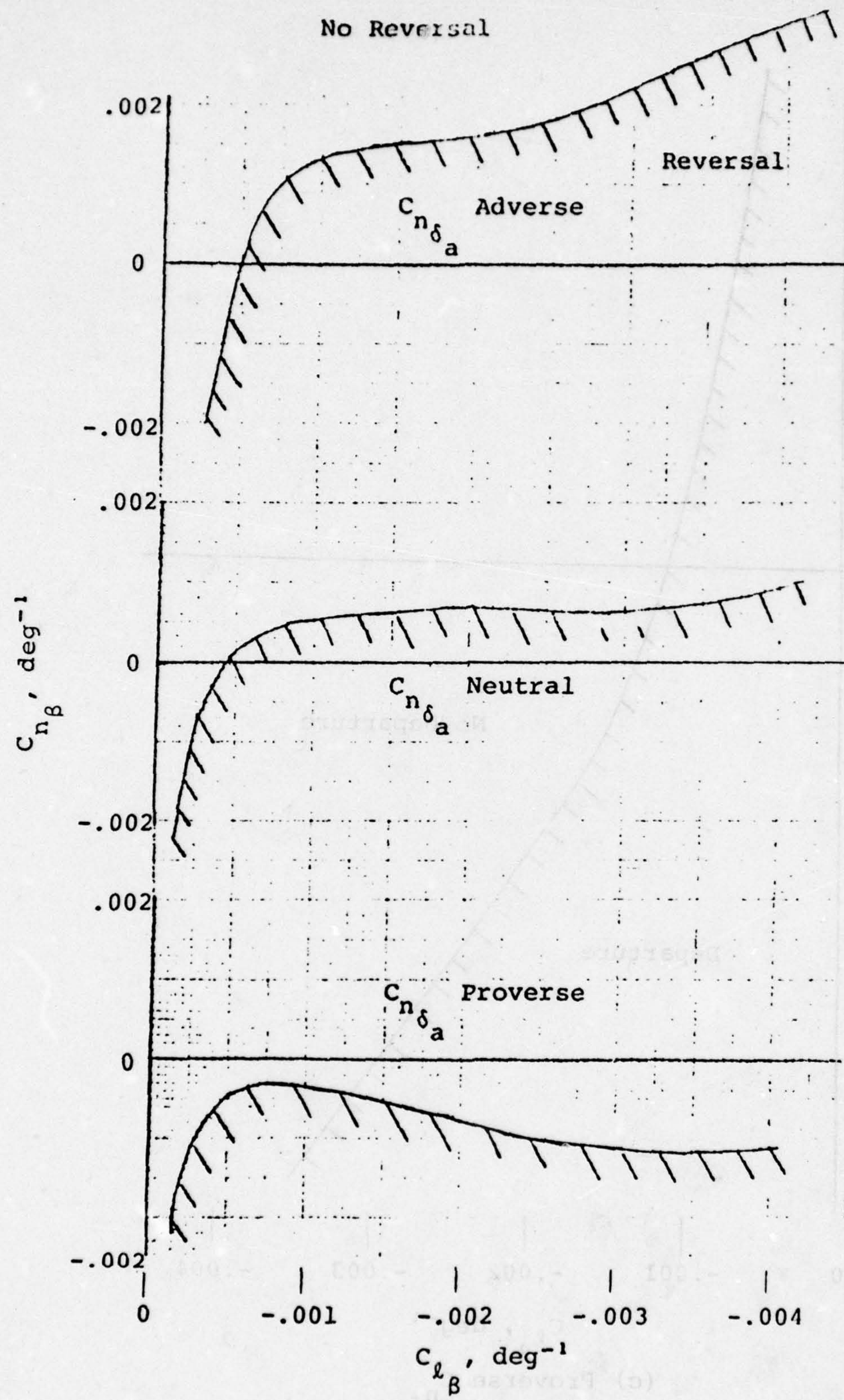
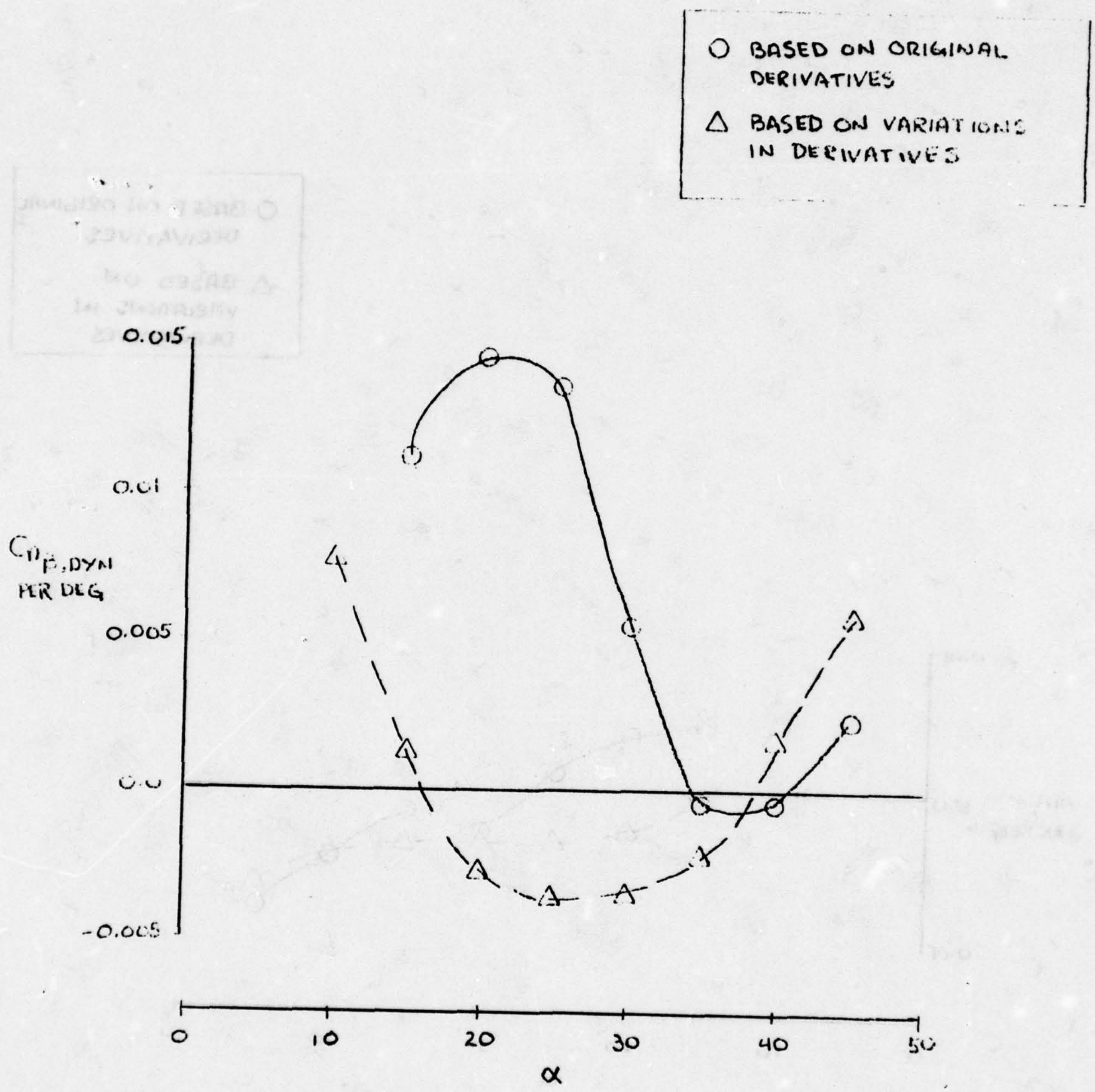
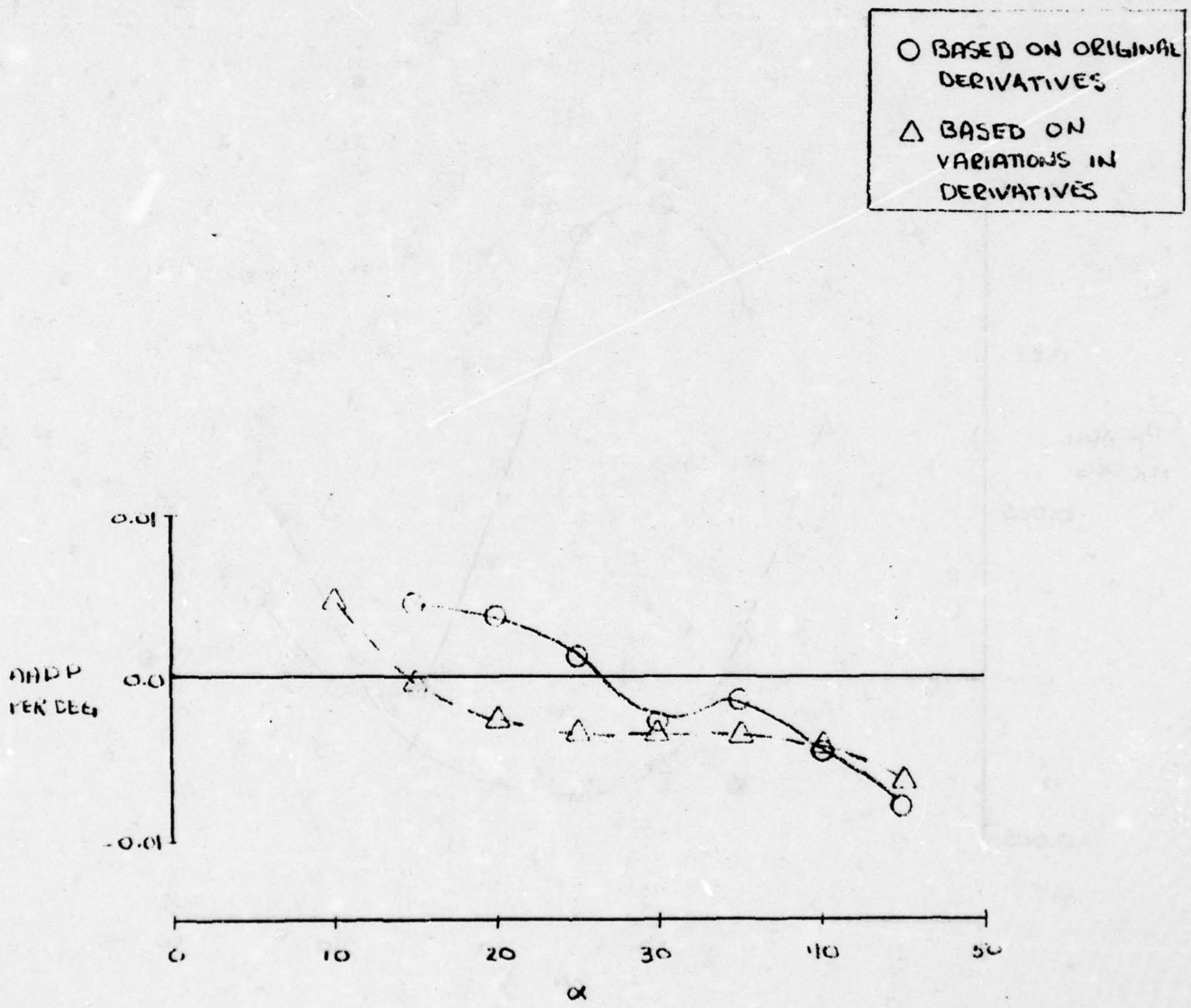


Figure 3 - Uncoordinated roll-reversal boundary. (From Ref. 6)



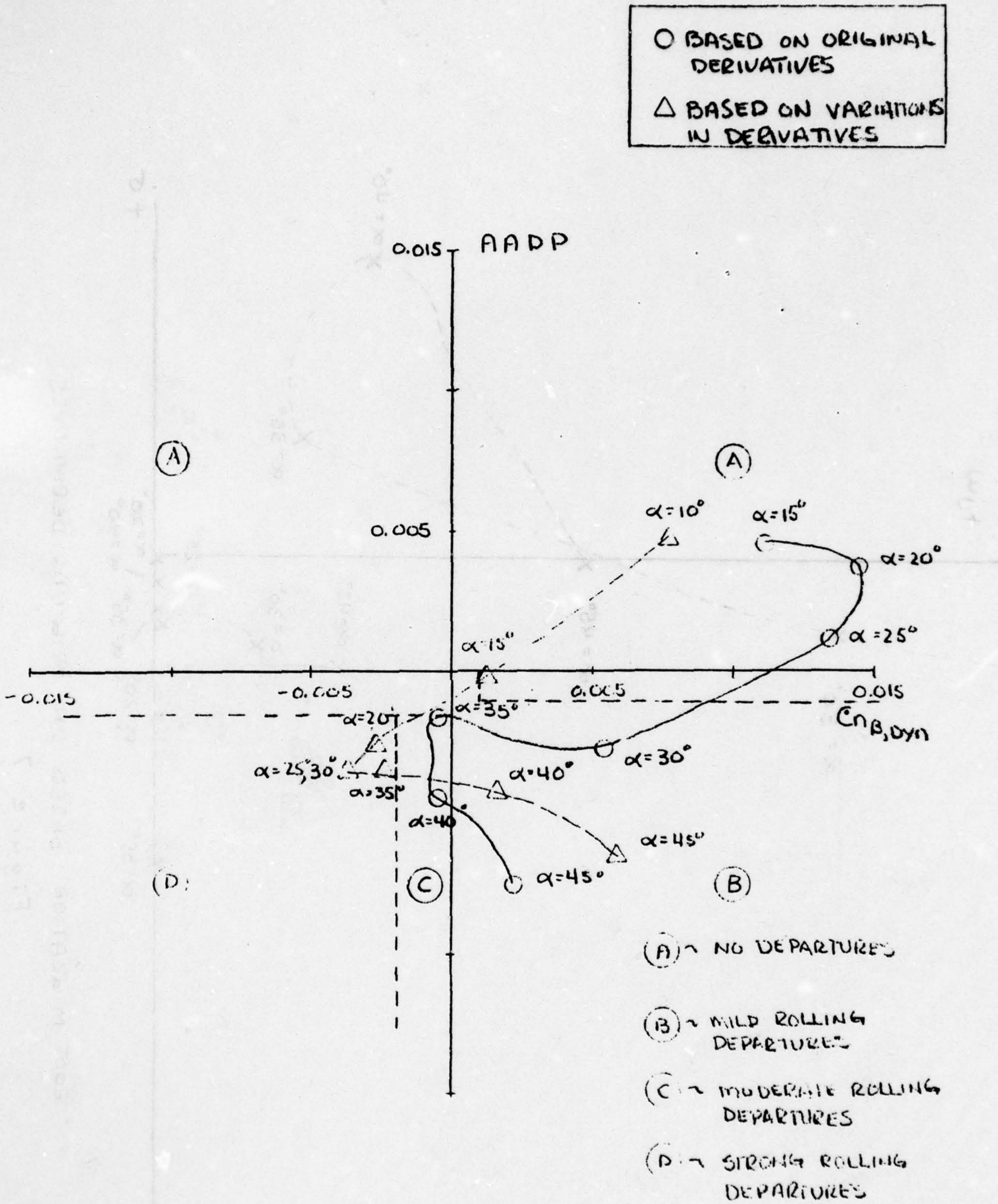
Variation of $C_{n_{\beta,DYN}}$ with angle of attack

Figure 4



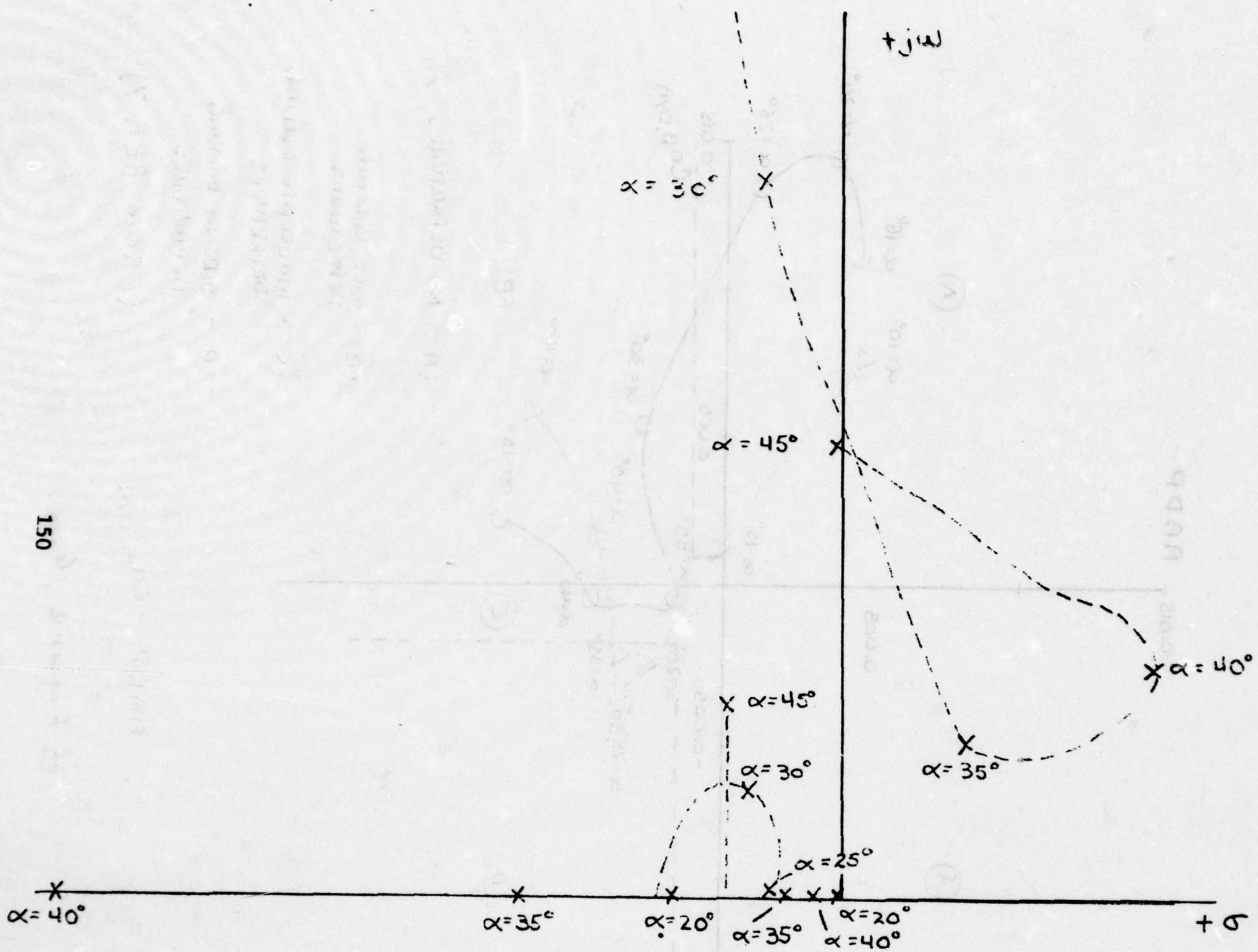
VARIATION OF AADP WITH ANGLE OF ATTACK

Figure 5



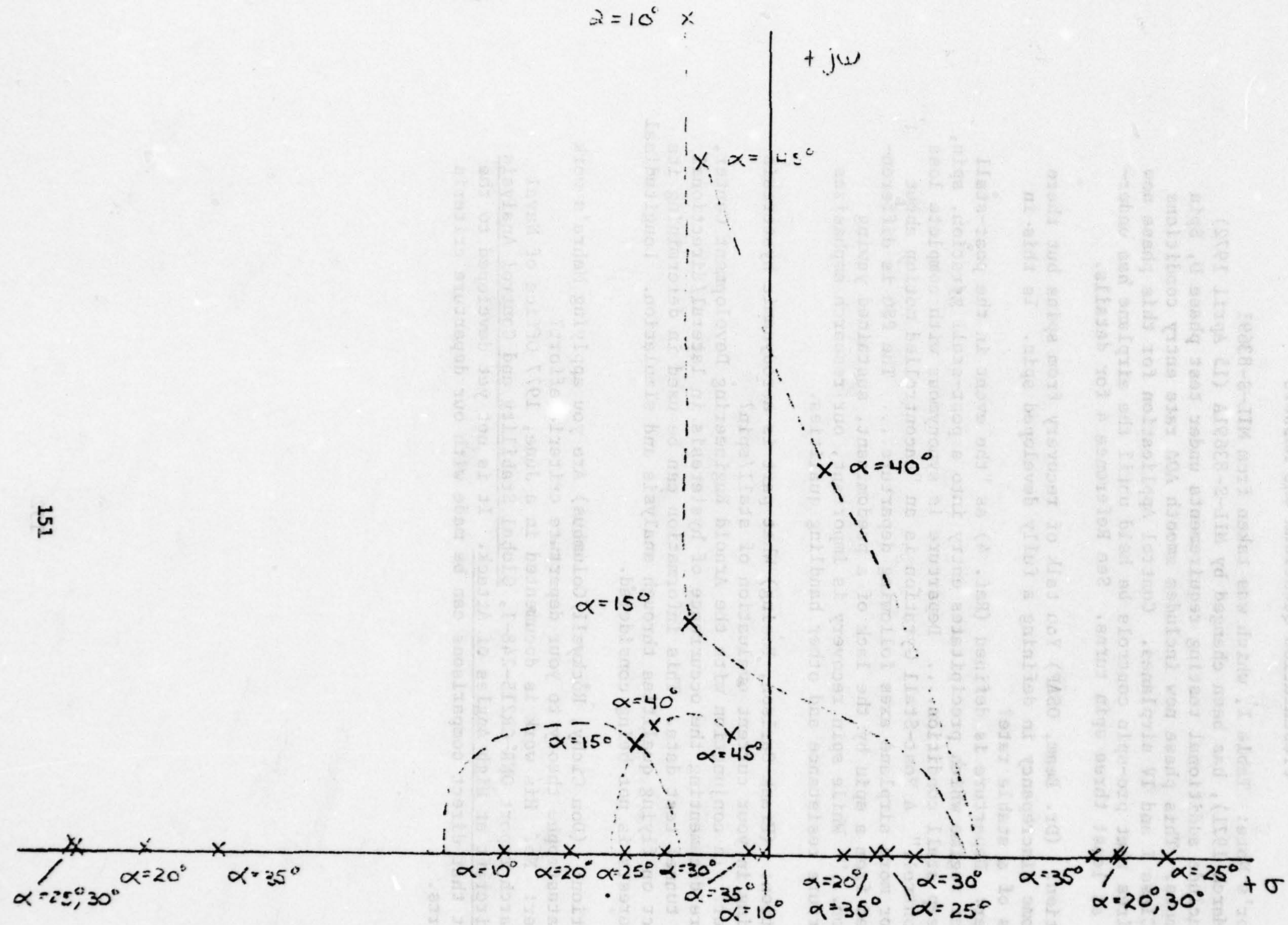
AADP - $Cn_{\beta, Dyn}$ (FROM REF. 7)

Figure 6



ROOT MIGRATION BASED ON ORIGINAL DERIVATIVES

Figure 7



ROOT MIGRATION BASED ON VARIATIONS IN DERIVATIVES

Figure 8

Presentation Questions and Answers

Author's Note: Table I, which was taken from MIL-S-83691 (31 March 1971), has been changed by MIL-S-83691A (15 April 1972) to include additional testing requirements under test phase D, Spin Attempts. This phase now includes smooth AOA rate entry conditions for Class I and IV airplanes. Control Application for this phase now requires that pro-spin controls be held until the airplane has undergone at least three spin turns. See Reference 4 for details.

Question: (Dr. Beam, OSAF) You talk of recovery from spins but there is some discrepancy in defining a fully developed spin. Is this in terms of a stable rate?
Answer: Departure is defined (Ref. 4) as "the event in the post-stall flight regime which precipitates entry into a post-stall gyration, spin, or deep stall condition.... Departure is synonymous with complete loss of control." A Post-Stall Gyration is an "uncontrolled motion about one or more airplane axes following departure.... The PSG is differentiated from a spin by the lack of a predominant, sustained yawing motion..." While spin recovery is important, our research emphasizes departure resistance and other handling qualities.

Question: (Frank Carlson, Boeing) What part is aerodynamic hysteresis playing in your current evaluation of stall/spin?
Answer: In conjunction with the Arnold Engineering Development Center, we are documenting the occurrence of hysteresis in lateral/directional wind tunnel test data. This information can be used in determining its effect on flying qualities through analysis and simulation. Longitudinal hysteresis is not being considered.

Question: (Don Cichy, Rockwell/Columbus) Are you applying Mehra's work in catastrophe theory to your departure criteria effort?
Answer: No. His work is documented in a June, 1977 Office of Naval Research report ONR-CR215-248-1, Global Stability and Control Analysis of Aircraft at High Angles of Attack. It is not yet developed to the point that direct comparisons can be made with our departure criteria efforts.

SUMMARY OF RECENT HANDLING QUALITY RESULTS AT STI PERTINENT TO REVISION OF MIL-F-8785B

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INTRODUCTION

The results of a number of handling quality programs which include analysis simulation and flight test are presented. Lessons learned from a series of moving base simulations involving approach and landing of STOL airplanes are summarized. This is followed by a review of the effects of large windshears. The importance of including such effects in the specification requirements is discussed. A method for predicting an overall pilot rating given separate ratings for each individual axis of control is presented. Finally some pilot describing function results which have application for defining equivalent system for path control are presented.

SOME CONSIDERATIONS FOR IDENTIFYING LONGITUDINAL PATH CONTROL DEFICIENCIES DURING APPROACH AND LANDING

A series of simulations were conducted to obtain the technical data required to assist the FAA in establishing certification criteria for STOL aircraft. For the most part these were conducted on the NASA Ames FSAA simulator.

A summary of the factors which tended to reveal vehicle deficiencies related to piloted control of flight path is given as follows:

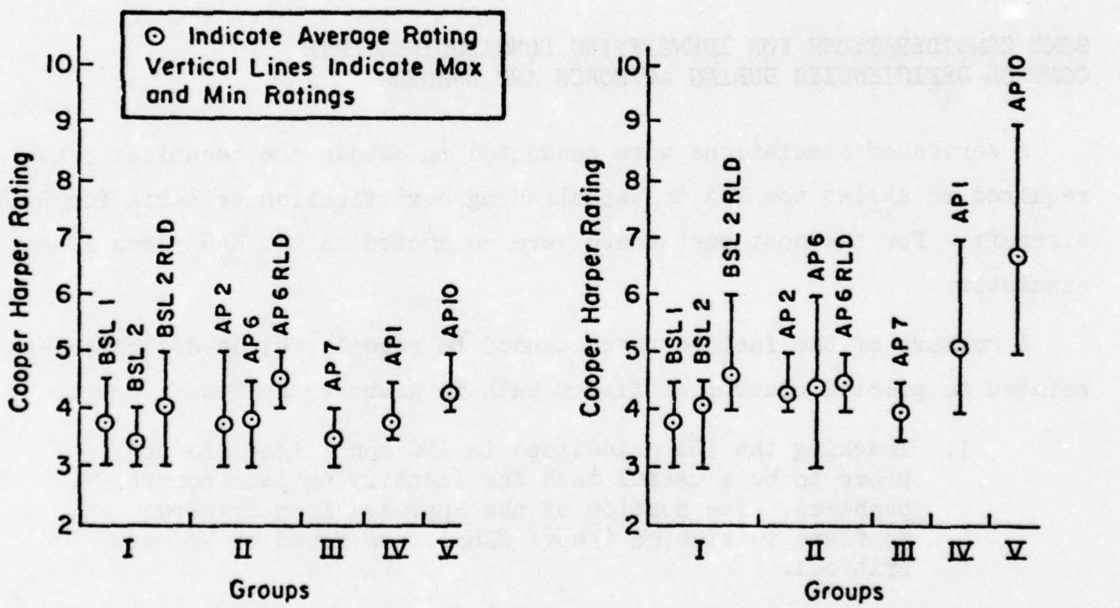
1. Tracking the ILS glideslope in IMC conditions did <u>not</u> prove to be a useful task for identifying path control problems. The portion of the approach from breakout to flare initiation (short final) was found to be most critical.

2. Even on short final, path control deficiencies were not always apparent in calm air. The addition of random turbulence (σ_{u_g} = 4.5 ft/sec) was found to be a key factor in separating out vehicles with severe path control deficiencies.

3. Vehicles with more subtle path control deficiencies were identified by introducing large discrete windshears just prior to touchdown.

4. Investigation of different random turbulence models revealed that in the critical region (short final), the dominant features of the turbulence are essentially identical. Hence, the identified vehicle deficiencies were not dependent on the turbulence model used.

An illustration which supports the first of the above conclusions is presented in Fig. 1. These data are taken from Ref. 1 which was a study to identify minimum acceptable manual STOL flight path control characteristics. The configurations range from fair to very bad. The data in Fig. 1a are for ILS tracking only (σ_{ug} = 4.5 ft/sec) with the runs being terminated at breakout (300 feet altitude). The pilot ratings show that this task did not highlight path control deficiences in any of the configurations. Figure 1b illustrates that when the task was expanded to include



a) Pilot Rating for Glideslope Tracking Task (IFR Only)

b) Pilot Ratings for Path Control Part of Composite Task (IFR and VFR Path Control)

Figure 1. Pilot Ratings for Path Control

visual lineup, flare and touchdown, certain of the configurations were found to be unacceptable. The pilot commentary, summarized in Table 1, verifies that path control deficiencies with the poor configurations became apparent when attempting to get set up for the flare in the presence of turbulence. Based on the pilot commentary (Table 1), the poor configurations are identified as much by the rating variability (5 to 9, 4 to 7, etc.) as by the absolute rating, itself. The rating variability appears to occur because of real changes in landing performance or pilot effort due to critical but random combinations of particular turbulence inputs and off-nominal flight; i.e., the same configuration can be very bad on some trials and acceptable on others. Unfortunately, the bad cases often look like pilot error and may not always be accounted for in the ratings.

EFFECT OF DIFFERENT TURBULENCE MODELS

Because, as just discussed, turbulence on short final is a critical factor in identifying vehicle deficiencies, the sensitivity of pilot opinion to the turbulence model deserves consideration and clarification. To this end, an analysis and simulation study was undertaken (Ref. 2) to identify the effect of different turbulence models on the critical last few hundred feet of the approach.

Figure 2 presents the calculated effect of the horizontal and vertical components of the MIL-F-8785B Dryden model on the RMS sink rate of a DHC-6 Twin Otter aircraft if attitude and throttle are fixed. Sink rate excursions due to the vertical component are seen to approach zero near touchdown. This is a consequence of the fact that scale length, L_w, and altitude approach zero simultaneously ($L_w = h$). The key implication of Fig. 2 is that the important variable for comparisons of different gust models is the variation of the horizontal component at speeds and altitudes consistent with short final. Figure 3 presents the normalized RMS sink rate excursions vs. the breakpoint of the horizontal gust filter for a simulated DHC-6 Twin Otter with altitude and throttle fixed. It shows that 1) maximum path disturbance occurs for values of V/L_u between $1/T_{\theta_1}$ and $1/T_{\theta_2}$, and 2) the sink rate disturbances for these values of V/L_u are relatively constant. Investigation of four

TABLE 1. PILOT COMMENTARY WHERE FLIGHT PATH CONTROL PROBLEMS ON SHORT FINAL WERE SPECIFICALLY NOTED (TASKS 2.1 AND 3.1)

	PILOT 1	PILOT 2	PILOT 5	PILOT 7	PILOT 8	PILOT 9
BSL1	None	Poor vertical speed response makes it easy to overcontrol Put on too much power to correct for a low condition and then don't get it off in time, etc.		None	None	Am flying glide slope (ILS) to get to window for flare
BSL2	None	None		I am having quite a bit of problems with the turbulence particularly during the final glide slope tracking and the flare	None	None
BSL2RLD	Requires moderate compensation on throttles to set up for flare			None		Poor sink rate to throttle response is responsible for problems in getting set up at flare point Flying IVSI to throttles even in close
AP1	The primary deficiency is a very sluggish sink rate to throttle response. The major problem is the inability to recover from off nominal vertical position in time to set up for landing on this short runway		Pilot rating is a 3 down to breakout and then a 7 on short final	The workload gets too high trying to get the power set for your flare, particularly with these last minute flight path corrections where the power can be going up and down	... real dicey to get a good sink rate and a good aim point on the runway	Primary difficulty was the considerable lag in the throttle and if your effecting a change on glide path the resulting change in sink rate late in the approach will give you real problems
AP2	The primary problem in landing is setting up for the flare with power in the presence of these fairly large gust disturbances	Recovery from turbulence effects coming into the flare was difficult		Turbulence is not a problem and getting set up for flare is also not a problem with this configuration		
AP6	None	None		None		
AP6RLD	Moderate compensation on sink rate control with power is required to set up the flare point			Sink rate response to attitude and power are good		None
AP7	None			None		None
AP10	The sluggish sink rate to throttle makes it difficult to get set up. My primary objection to this configuration lies in the inability to control sink rate during the last several hundred feet of the approach			The main problem with flight path control is that flight path angle washes out after a throttle input. This problem is especially noticeable as you approach the flare point and even during the flare	Got low and slow, a bear to correct	Seems very sensitive to throttle making it difficult to set up for flares. Extremely hard to get into proper flare window

Notes: Blank space means pilot did not fly the configuration.
"None" means that no specific comments relative to flight path control on short final were recorded.

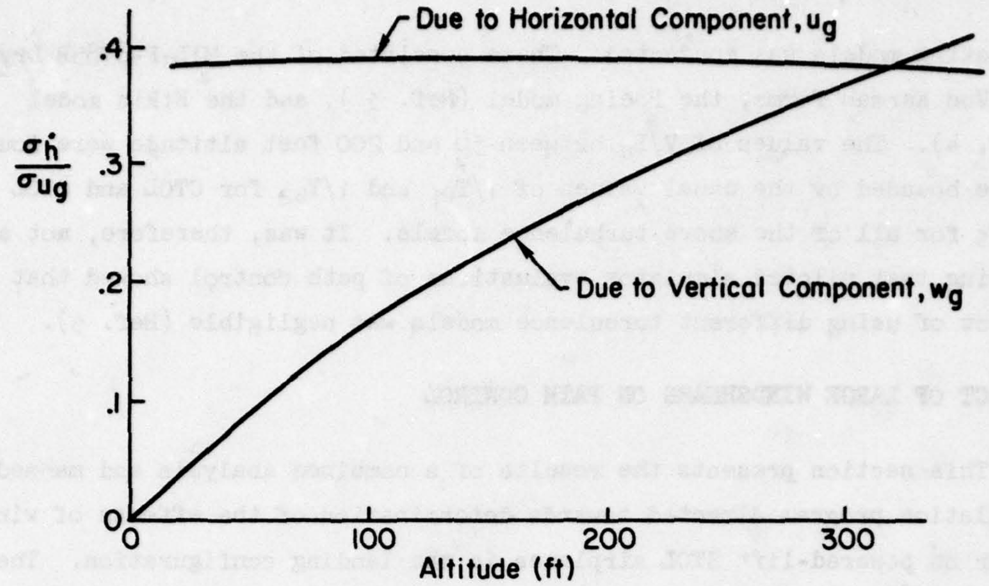


Figure 2. Effect of Horizontal and Vertical Gusts for Varying Altitude (Attitude and Throttle Fixed)

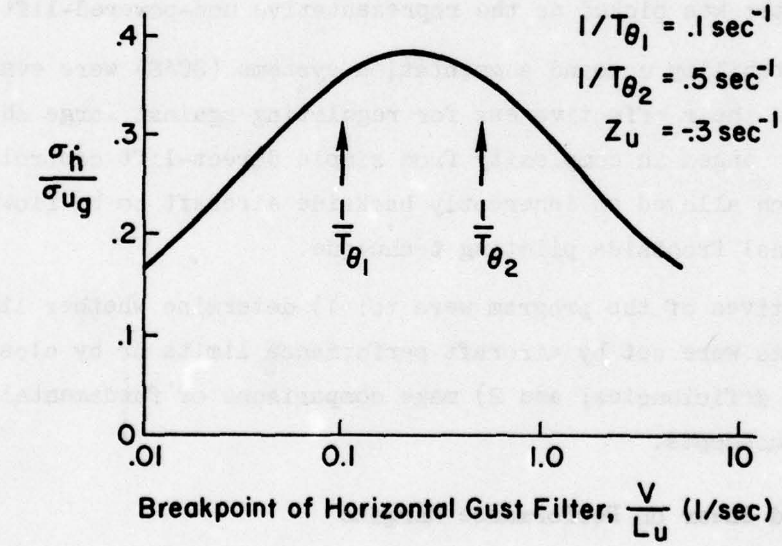


Figure 3. Effect of Horizontal Gust Filter Breakpoint on Altitude Rate Dispersion

competing models was conducted. These consisted of the MIL-F-8785B Dryden and Von Karman forms, the Boeing model (Ref. 3), and the Etkin model (Ref. 4). The values of V/L_u between 50 and 200 feet altitude were found to be bounded by the usual values of $1/T_{\theta_1}$ and $1/T_{\theta_2}$ for CTOL and STOL aircraft for all of the above turbulence models. It was, therefore, not surprising that piloted simulator evaluations of path control showed that the effect of using different turbulence models was negligible (Ref. 5).

EFFECT OF LARGE WINDSHEARS ON PATH CONTROL

This section presents the results of a combined analysis and manned simulation program directed towards determination of the effects of wind shear on powered-lift STOL airplanes in the landing configuration. The powered-lift concepts considered were the externally blown flap (EBF), upper surface blowing (USB), and the Augmentor Wing. Descriptions of these powered-lift STOL concepts may be found in Refs. 6-8, respectively. A non-powered-lift shorthaul concept was also considered in order to provide a basis for comparison for evaluation of accident potential. The De Havilland DHC-6 Twin Otter was picked as the representative non-powered-lift STOL.

Several stability command augmentation systems (SCAS) were evaluated to investigate their effectiveness for regulating against large shears. These systems ranged in complexity from simple direct-lift control (DLC) to a SCAS which allowed an inherently backside aircraft to be flown using the conventional frontside piloting technique.

The objectives of the program were to: 1) determine whether limiting characteristics were set by aircraft performance limits or by closed-loop pilot/vehicle deficiencies; and 2) make comparisons of fundamental STOL augmentation concepts.

Effect of Wind Shear on Performance Margins

The primary effect of wind shear is to change airspeed. This results in excursions from the flight path due to the corresponding change in lift. Hence regulation against wind shear can be accomplished by changing the lift or by accelerating the aircraft so that the net airspeed change is minimized. Physical interpretation of the acceleration required to cancel the

effect of wind shear is simplified if it is treated as an "effective flight path angle," e.g., $a_x = g\gamma_{eff}$. The condition for maintaining constant airspeed in a wind shear is expressed in terms of γ_{eff} as follows:

$$\gamma_{eff} = \gamma_i \left(1 + \frac{V_w}{V_a}\right) + \frac{1}{g} \dot{V}_w \tag{1}$$

where γ_i = inertial flight path angle (glide slope angle), and $\sin \gamma \doteq \gamma$ and $\gamma_a \doteq \gamma_i[1 + (V_w/V_a)]$. In order to keep the sign convention for winds consistent with the usual formulation of the equations of motion, a positive wind has been defined as a tailwind.

γ_{eff} is a fictitious flight path angle used to define a speed/power equilibrium point on the usual γ-V representation. This point represents the required acceleration/deceleration capability to regulate against wind and wind shear in terms of flight path angle capability in calm air.

The aircraft performance capability may be compared to the performance required to maintain zero glide slope error in wind and wind shear by comparing γ_{eff} with the maximum or minimum achievable γ on a γ-V plot. This is illustrated in the generic sketch shown in Fig. 4 (γ-V shapes typical of an EBF or USB STOL concept). This sketch is indicative of the effects of a large steady headwind which is shearing towards zero (effects of negative wind and positive wind shear are additive). The effective flight path angle is a function of the wind speed, V_w, and therefore changes during the time the airplane is in the wind shear as follows:

$$\gamma_{eff} = \gamma_i \left(1 + \frac{V_w}{V_a}\right) + \frac{\dot{V}_w}{g} + \gamma_i \frac{\dot{V}_w}{V_a} t \tag{2}$$

Thus, for the usual case where wind is decreasing during the approach, a given wind shear may initially exceed the aircraft control power ($\gamma_{max} < \gamma_{eff}$) until the steady component of wind decreases sufficiently to allow control, as illustrated in Fig. 5. It therefore seems logical to define the limiting combinations of steady wind and wind shear when $\gamma_{eff} = \gamma_{max}$ at t = 0, e.g., flight path control margin equals zero at the beginning of the shear. This

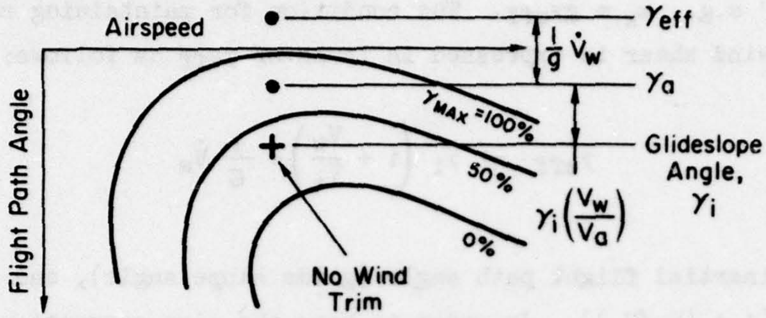


Figure 4. Effect of Wind and Wind Shear on Performance Margins

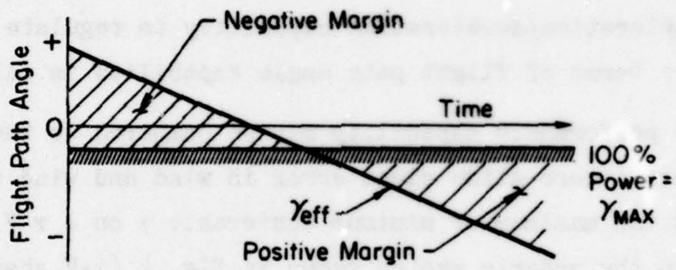


Figure 5. Illustration of Change in γ_{eff} with Time in a Decreasing Headwind Shear

is done by substituting γ_{max} and γ_{min} for γ_{eff} in Eq. (1) and solving for limiting values of wind shear, $\dot{V}_w$, e.g.,

$$\dot{V}_w = g\left[\gamma_{\substack{max\\(min)}} - \gamma_i\left(1 + \frac{V_w}{V_a}\right)\right] = g\left[\gamma_{\substack{max\\(min)}} - \gamma_a\right] \quad (3)$$

The boundaries which derive from Eq. (3) are plotted in Fig. 6 where γ_{max} was taken as zero and γ_{min} as −10 deg. These numbers were picked as a consequence of the tentative STOL airworthiness requirements which dictate a capability of 4 deg below the glide path and level flight in the up direction. For decreasing winds (second and fourt quadrant) the path control

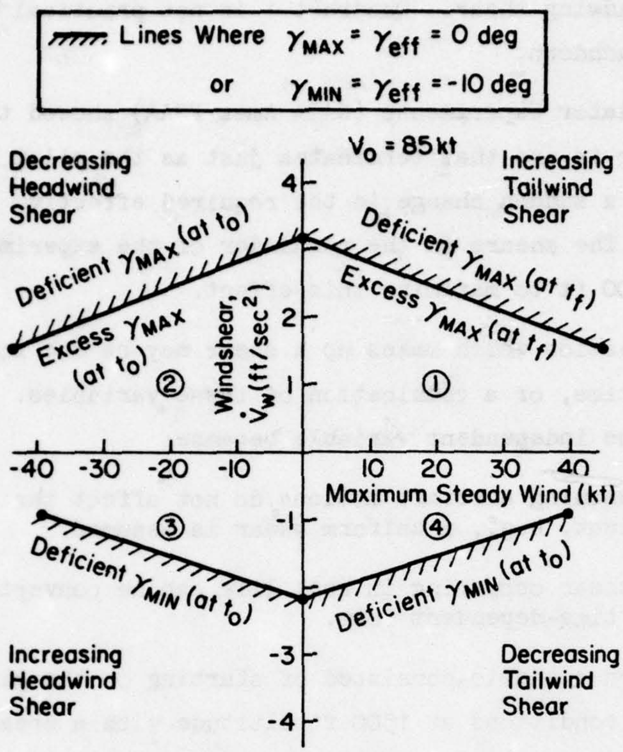


Figure 6. Boundaries Where Flight Path Control Margin Equals Zero ($\gamma_{\max(\min)} = \gamma_{eff}$)

margin is zero ($\gamma_{eff} = \gamma_{max}$) when the shear starts and is positive ($\gamma_{eff} < \gamma_{max}$) for the remainder of the shear. For increasing winds the path control margin is initially positive ($\gamma_{eff} > \gamma_{max}$) and degrades to zero when the shear ends ($t = t_f$).

Simulation Test Matrix

The matrix of discrete wind shears used in the simulation program was developed to include combinations of steady wind and wind shear on both sides of the theoretical performance boundaries developed in Eq. (3) and plotted in Fig. 6. The test matrix primarily concentrated on Quadrants 2 and 4 with some runs in Quadrant 3. Quadrant 3 proved to be less critical

because of the low groundspeed and favorable effect on lift in the flare of an increasing headwing shear. Quadrant 1 is not practical because it implies a tailwind at touchdown.

Initial simulator experiments (NASA Ames FSAA) showed that the most critical type of shear is one that terminates just as the pilot is coming into the flare due to a sudden change in the required effective flight path angle ($\Delta\gamma_{eff} = \dot{V}_w/g$). The shears in the remainder of the experiment were terminated between 50 and 100 ft to maximize this effect.

The wind variation which makes up a shear may be due to changes in altitude, position, time, or a combination of these variables. It was decided to use time as the independent variable because:

- The ensuing aircraft motions do not affect the shear gradient, e.g., a uniform shear is assumed.
- Any shear occurring in real life can be converted to a time-dependent form.

The simulation scenario consisted of starting on the glide slope and localizer in IFR conditions at 1500 ft altitude with a breakout to VFR at 300 ft at which point the pilot visually acquired the runway and continued the approach to a landing.

Experimental Results

A comparison of the pilot commentary with the accident potential rating scale from Ref. 9 was made to determine if a number could be associated with the pilots' opinion of unacceptable hazard. A reasonably consistent trend was identified which correlated commentary relating to unacceptable hazard and an accident potential rating of 4.

The results of the piloted simulator program are summarized in Fig. 7 by fairing approximate pilot rating boundaries where the accident potential rating was equal to 4 on a grid of steady wind vs. wind shear. The separation between these pilot rating boundaries and the theoretical boundaries [defined by Eq. (3) and plotted in Fig. 6] is a measure of shear vulnerability. That is, when the pilot rating boundary lies below the theoretical boundary in Fig. 7, the configuration tends to be highly vulnerable to decreasing headwind shears.

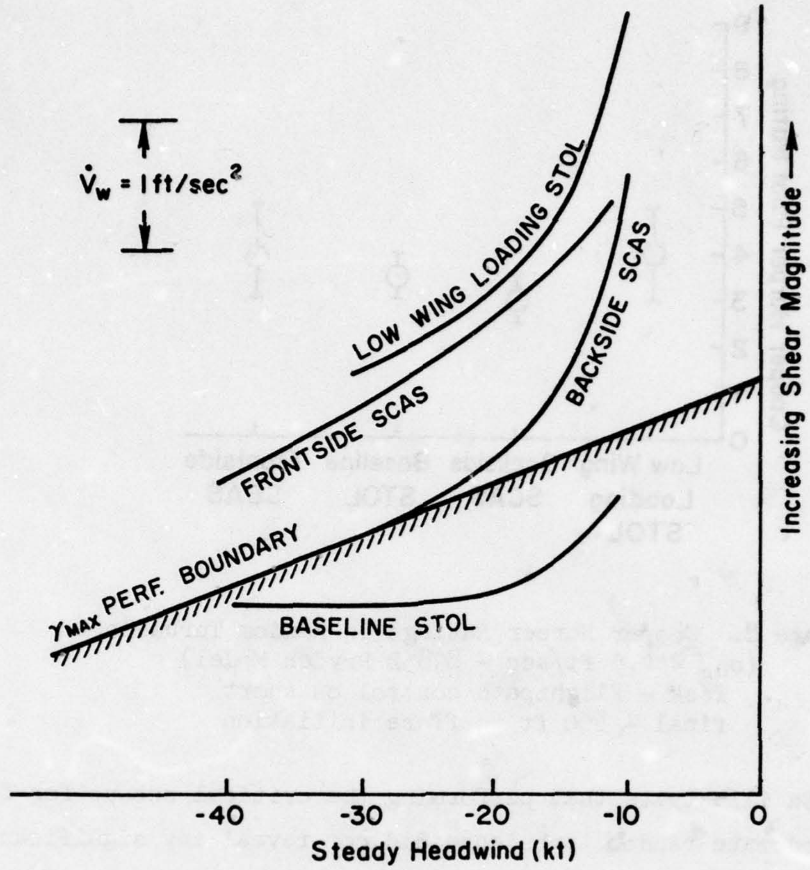


Figure 7. Comparison of Unacceptable Accident Potential Rating with γ_{max} Performance Boundary (Decreasing Headwind Shear)

Each of the configurations in Fig. 7 are discussed in detail in Ref. 10 along with an analysis of vehicle deficiencies which relate to flight path control in windshear. The point of emphasis in the present paper, however, is that the use of a critically timed discrete windshear separates out the more subtle path control deficiencies. This is well illustrated by comparison of the hazard boundaries in Fig. 7 and the Cooper Harper Ratings for these same configurations with a 4.5 ft/sec RMS random turbulence level in Fig. 8.

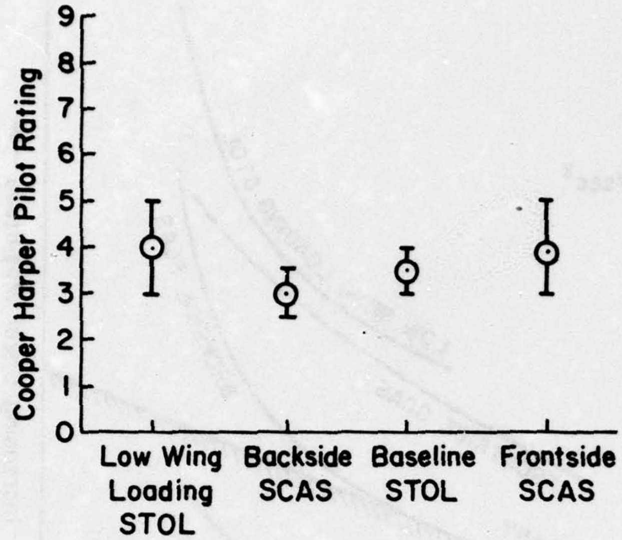


Figure 8. Cooper Harper Ratings in Random Turbulence (σ_{u_g} = 4.5 ft/sec - 8785B Dryden Model)
Task - Flightpath control on short final - 300 ft to flare initiation

The Fig. 8 data illustrate that performing the critical set-up-for-flare maneuver in moderate random turbulence did not reveal any significant differences between path regulation among the tested vehicles. Figure 7, however, shows the Baseline STOL to be highly vulnerable to decreasing headwind shears. Some improvement is shown when the Baseline STOL is augmented with a washed out throttle to spoiler crossfeed (Backside SCAS). The Frontside SCAS and Low Wing Loading STOL are seen to be the least vulnerable to large discrete shears.

Conclusion

The use of a random turbulence model does not generally allow detection of path control deficiencies that may be critical to safety in a large winshear on short final. There should be some provision in the MIL-F-8785B standard that will insure adequate performance in a critically timed discrete shear. The exact form that such a requirement should take is not clear at

this time. Some research is required to consolidate the data from the windshear programs that have been completed to date and to develop a tentative criterion.

PREDICTION OF OVERALL PILOT RATINGS FROM SINGLE AXIS RATING DATA

An empirical formula for estimating the combined effect of pilot ratings for m individual control axes was developed in Ref. 11 and is presented below:

$$R_m = 10 + \frac{1}{(8.3)^{m-1}} \prod^{m} (R_i - 10) \tag{4}$$

Where R_i is the rating for each individual axis.

Correlations obtained between actual and predicted pilot rating data are given in Fig. 9. Here the ordinate is the multiple-axis rating computed, from the observed single-axis ratings, R_i, using Eq. 4. The abscissa, of course, is the actual multiple-axis rating for the same set of individual axis ratings. The spread in the individual points is due to the uncertainties in either or both the single-axis and multi-axis rating data (e.g., rating = 4-1/2 to 5). The Ref. 12 data are most pronounced in this respect, the spread here reflecting the differences in single-axis ratings delivered before and after the 3-axis runs, and also differences between the first and second series of 3-axis runs, themselves; the center point is based on the overall averaging of the single and 3-axis data given in Ref. 12.

The correlation shown is on the whole quite good for the region of most interest, i.e., ratings between 2 and 7. In fact for this region, and neglecting the spreads shown for the Ref. 12 data, the computed rating agrees with the observed rating within about half a point.

Implications for MIL-F-8785B(ASG) Revision

The impact of the foregoing on the single-axis requirement to achieve various levels of multiple-axis (i.e., whole task) flying qualities is potentially quite drastic. That is, for Level 1 whole task flying qualities

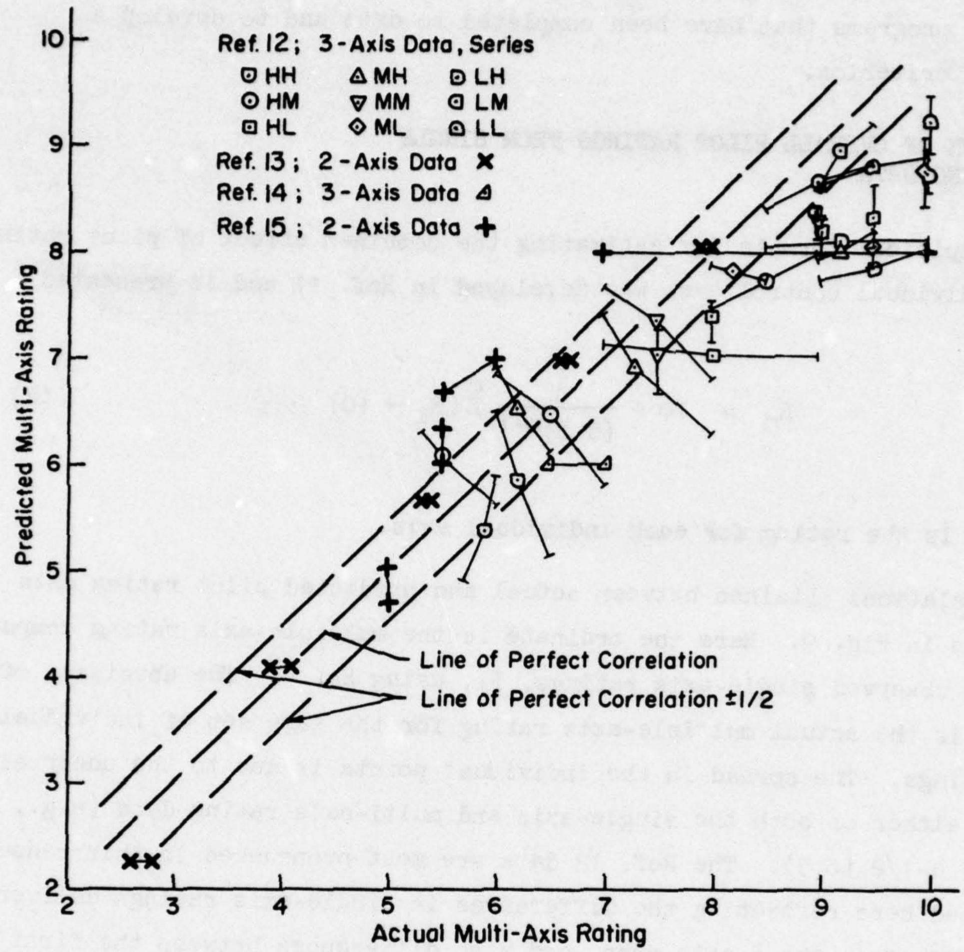


Figure 9. Correlations Obtained Using Equation 4.

corresponding to a multiple-axis rating of 3.5 or better (Ref. 16), the required longitudinal and lateral-directional flying qualities must be better, according to Eq. 4, than about $2.65 + \Delta$ for one and $2.65 - \Delta$ for the other (where $2.65 + \Delta < 3.5$). Taking the most beneficial view of the 1/2 rating point "inaccuracy" (Fig. 9) of Eq. 4 increases these values to $2.95 + \Delta$ and $2.95 - \Delta$ (where $2.95 + \Delta < 4$). Such an explicit requirement for longitudinal and lateral-directional flying qualities, which are each a little <u>better</u> than the Level 1 (3.5) boundary is somewhat in keeping with vague undocumented "stories" of aircraft which were not satisfactory because

too many of their parameters, while each individually satisfactory, were very near the boundary value (a possibility mentioned also in Ref. 16, Item 1.5).

On the other hand, it must be recognized that the data used (Ref. 16) to establish the various level boundaries were, wherever possible, based on the results of flight-test investigations where tasks other than those being rated were necessarily present. Since such other tasks (or parameters) were supposedly in the "good" (Level 1) region, it seems pertinent to consider that for a configuration rated 3-4 the pilot may have been flying longitudinal and lateral axes each rated about 3. Past experience with preceding versions of the MIL Spec tend to further support the above observation; that is, airplanes near but within the satisfactory boundary values in both longitudinal and lateral-directional handling are generally satisfactory overall.

Because of the above considerations and the lack of definitive in-flight data on multiple-axis effects, it seems inadvisable to alter the Level 1 definition. However it is still appropriate considering the evidence herein, to issue a warning requiring further explicit study of those situations where both longitudinal and lateral-directional flying qualities approach very near the Level 1 boundaries.

The Level 2 boundaries are also based on the practice of "good" (i.e., Level 1) remaining parameters. This means that if one axis of the airplane is worse than Level 1 but better than, or equal to, the Level 2 boundary the other axis must be better than or equal to the Level 1 boundary. This interpretation of the actual data used to establish the Level 2 boundaries is hinted at in Ref. 16, where it is noted that some of the Level 2 boundaries were somewhat arbitrarily "stiffened" so that two axes in the Level 2 region might still represent Level 2 conditions. The actual "stiffening" required to produce this state of affairs is quite extreme, based on the present findings, and it seems doubtful that such extreme stiffening was actually or uniformly applied to the data.

Analysis in Ref. 11 suggests that, neglecting whatever "stiffening" was applied to "some" requirements (Ref. 16), a proper Level 2 definition reflects conditions where:

a. either the longitudinal or lateral-directional axis is better than, or equal to, Level 1; and the other axis is worse than Level 1 but no worse than the Level 2 boundary, or

b. both the longitudinal and lateral-directional axes are worse than Level 1 but no worse than a boundary halfway between the Level 1 and Level 2 boundaries.

SOME EXPERIMENTAL DATA ON EQUIVALENT SYSTEM FORMS FOR PATH CONTROL

The equivalent system approach to the specification of handling qualities has shown considerable promise (For example, see Ref. 17). However, all the work accomplished to date has involved attitude control. This section presents some moving base simulator data which suggests an equivalent system form for path control when operating on the backside of the power required curve e.g., pilot controls airspeed with pitch attitude and sink rate with power.

A review of the comments for AP1 and AP10 in Table 1 reveals that the pilots had a very difficult time trying to sort out what the actual problem was. Some said that the response was sluggish, probably referring to the fact that the longer-term flight path correction was a lot less than indicated from the initial response. The engine lag was decreased from the nominal 1.5 sec to 0.5 sec for several pilots. All indicated that they could see the effect, but it was of no help in controlling flight path for these two configurations. (There was no change in pilot rating.) This served as evidence that the pilots were <u>not</u> referring to engine lag effects when commenting on the excessively "sluggish" response of AP1 and AP10.

The fundamental closed-loop piloting problem was analyzed using the frequency response characteristics of the sink rate, $\dot{h}$, to throttle, δ_T, transfer functions (for constrained attitude) plotted in Fig. 10 for AP1 (a bad configuration) and AP2 (a good configuration). Utilizing experimental and theoretical results from the theory of manual control (for example, see Ref. 18), it can be shown that the flat region in the frequency response (for AP1) represents a fundamental limitation on closed-loop control. This stems from the fact that the human operator <u>always</u> tries to adjust his control inputs so as to equalize the vehicle frequency

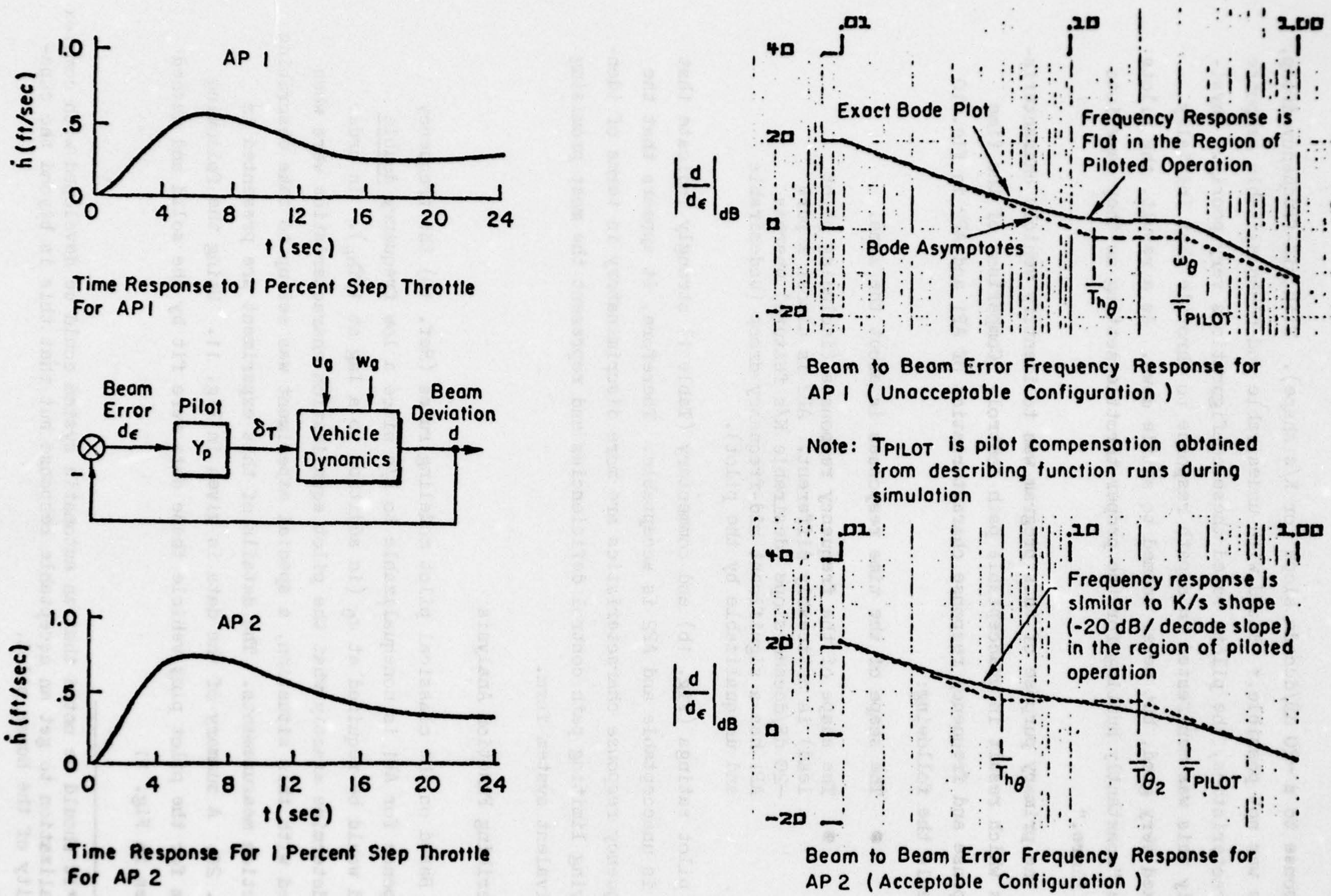


Figure 10. Time and Frequency Response Characteristics of an Acceptable and an Unacceptable Configuration

response to a −20 dB/decade slope (or K/s shape). With mid-frequency droop, this was not possible.* Faced with undesirable and nonequalizable response characteristics, the pilots rated these configurations very poorly. Physically this was manifested in a path response to throttle that initially looked very good, but never seemed to settle down. As a result, the pilots were "constantly hunting for the proper throttle setting as they came into the flare."

The primary purpose of this program was to identify vehicle characteristics which result in unacceptable path control. Comparison of the time response and frequency response characteristics of AP1 and AP2 in Fig. 10 reveals the following:

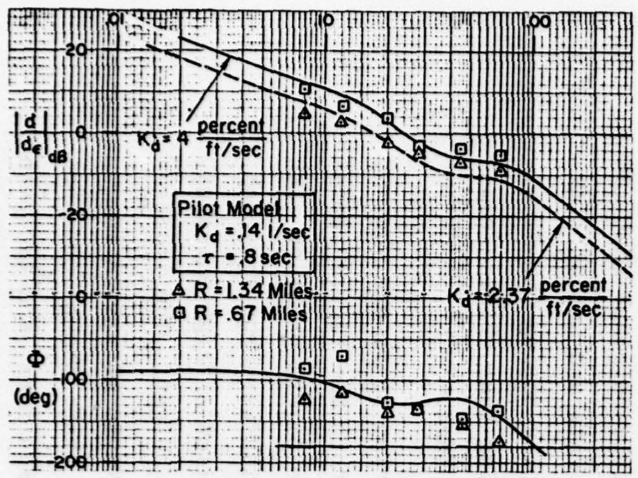
- The shape of the time responses is about the same.
- The shape of the frequency responses (including pilot lead) is noticeably different. AP2 is almost a pure −20 dB/decade slope (desirable K/s feature) whereas AP1 has a significant mid-frequency droop (undesirable and unequalizable by the pilot).

The pilot ratings (Fig. 1b) and commentary (Table 1) strongly indicate that AP1 is unacceptable and AP2 is acceptable. Therefore, it appears that the frequency response characteristics are more discriminatory in terms of identifying limiting path control deficiencies and represent the most promising equivalent system form.

Describing Function Analysis

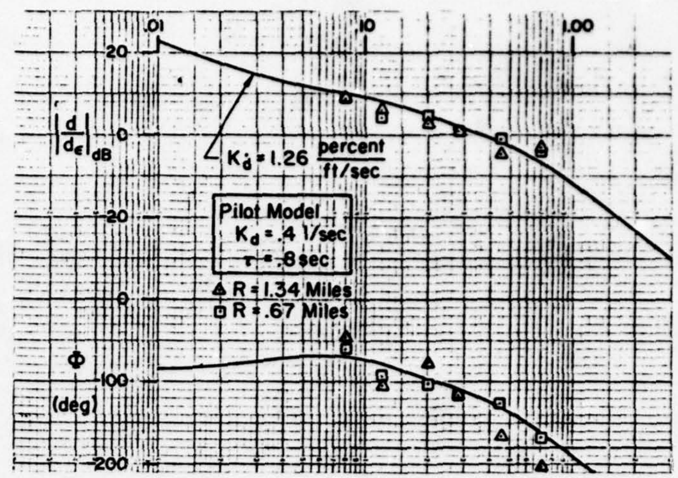
Based on the classical pilot modeling rules (Ref. 19) the frequency response for AP1 is nonequalizable to K/s since a low frequency double lead would be required at ω_θ (in addition to a lag at $1/T_{h_\theta}$). In order to determine exactly what the pilot equalization characteristics were when faced with this situation, a special experiment was set up to take describing function measurements. The details of this experiment are presented in Ref. 20. A summary of the data is given in Fig. 11. Using the following form for the pilot plus vehicle these data were fit by the solid and dashed lines in Fig. 11

*It should be noted that an automatic system could be developed with complex equalization to get an acceptable response but that this is beyond the capability of the human pilot.



a) Group I (BSL 2)

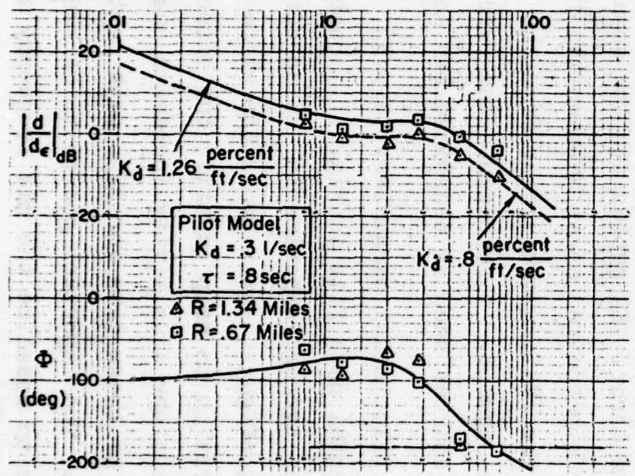
Pilot Rating 3-4½*



b) Group II (AP 6)

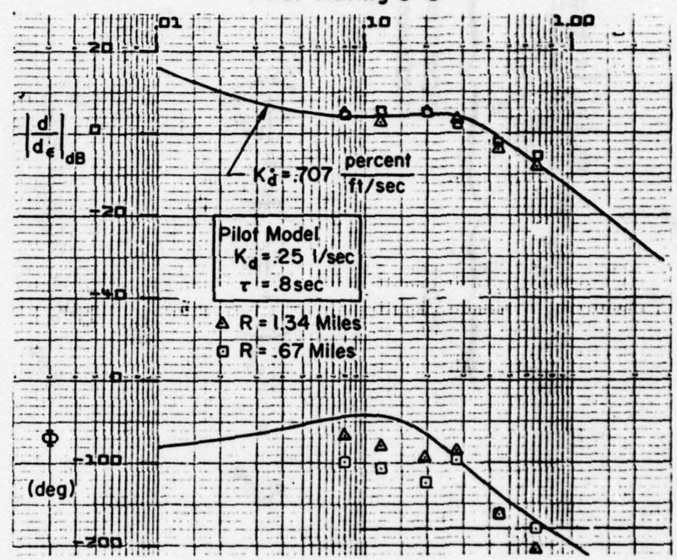
Pilot Rating 3-6

*Pilot ratings are for path control segment of composite task.



c) Group IV (AP1)

Pilot Rating 4-7



d) Group V (AP10)

Pilot Rating 5-9

Figure 11. Experimental-Analytical Pilot/Vehicle Model Correlations

$$\frac{d}{d_\epsilon} = Y_p Y_c \doteq \frac{Y_p N^\theta_{\delta_e} \delta^d_T}{N^\theta_{\delta_e}}$$

$$\doteq \underbrace{K_{\dot{d}}(s + K_d)e^{-\tau s}}_{\text{Pilot}} \underbrace{\frac{M_{\delta_e} Z_{\delta_T}\left(s + \frac{1}{T_{d\theta}}\right)}{s N^\theta_{\delta_e}\,(T_e s + 1)}}_{\text{Augmented Airframe}} \qquad (6)$$

where

$$N^\theta_{\delta_e} = M_{\delta_e}\left(s + \frac{1}{T_{\theta_1}}\right)\left(s + \frac{1}{T_{\theta_2}}\right) \qquad (7)$$

or

$$M_{\delta_e}(s^2 + 2\zeta_\theta \omega_\theta s + \omega_\theta^2) \qquad (8)$$

K_d represents the pilot's internally derived sink rate command based on a beam error. $K_{\dot{d}}$ represents the amount of throttle response that was used for a perceived error between the target sink rate and actual sink rate on the IVSI instrument or from the visual display during the final approach segment. τ represents the overall pilot lag that arises from several sources such as neuromuscular and scanning lags.

The pilot model parameters (K_d, $K_{\dot{d}}$, τ) were varied to obtain the experimental data fits in Fig. 11. Each data point in the figure represents the average experimental value across all the pilots who flew each of the configurations. Two glide slope sensitivities were run for each configuration to help quantify the effect of the pilot's "tightening up" as glide slope sensitivity increases near decision height. The implications of these data for STOL path control are covered in detail in Ref. 20. The important points to be made here are that: the pilots were not able to equalize the mid frequency shelf to a K/s form; and the path control pilot ratings (shown below each plot in Fig. 11) are significantly

degraded for those cases where the pilots were unable to equalize the effective controlled element to a K/s shape (AP1 and AP10).

QUESTIONS AND ANSWERS

1. Wayne Thor, ASD; What were magnitudes of shear? How about combinations of wind magnitude/shear?

   Wind shears varied from 1 ft/sec^2 to 6 ft/sec^2. The maximum steady wind was 40 kt (see Ref. 9 for more detailed answer).

2. D. Moorhouse: In the accidents at Logan and the 727 at Kennedy, the aircraft had the capability to survive the shears. Would a specification have helped this?

   It is difficult to answer this with any confidence, but I would suspect that a specification would at least make the vulnerability of these aircraft to windshears more apparent.

3. Bill Levison, BBN: Are factors that deal with the pilot's ability to detect and identify shears being considered in proposed revisions dealing with performance in shears?

   They were not considered in the work reported here. It was noted, however, that a sudden large change in airspeed was used as a cue.

4. Don West, Boeing: Was engine response time modeled?

   Yes. In fact the throttle spoiler crossfeed washout was set to inverse model the engine lag in the augmented configurations.

5. Dwight Schaeffer, Boeing: Was there any difference in the tailwinds?

   Not really. We also looked at decreasing tailwind shears which showed up as overshoots. The shear vulnerability remained about the same as shown on Fig. 7. These data are given in more detail in Refs. 9 and 10.

6. Chick Chalk, CALSPAN: Was the turbulence model really 8785B or modified?

   It was the 8785B model.

7. Bill Levison, BBN: Did you use any non-Gaussian turbulence models?

   No.

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FLYING QUALITY REQUIREMENTS FOR THE AMST

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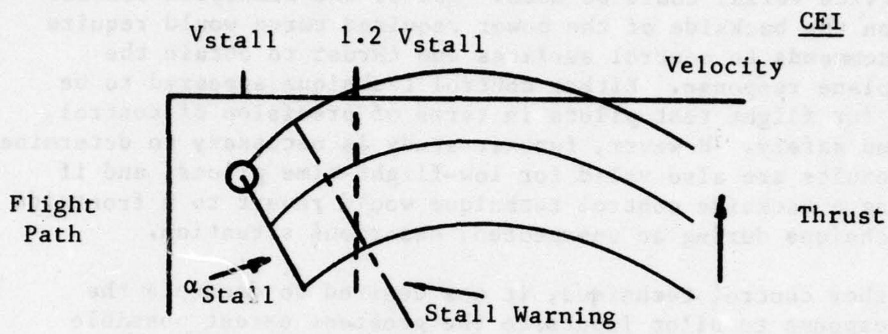
The major USAF flying qualities requirements for the production Advanced Medium STOL Transport (AMST) are discussed. The Military Specification for Flying Qualities of Piloted Airplanes (MIL-F-8785B, Reference 1) was modified by the Engineering Office of the AMST System Program Office (SPO) to account for powered lift and Short Takeoff and Landing (STOL) operation. These modifications were discussed thoroughly with the two AMST prototype contractors, various agencies of the USAF, including the Military Airlift Command, and the National Aeronautics and Space Administration (NASA). The primary tactical mission of the AMST is to carry a 27,000 pound payload 400 nautical miles into a 2100 foot long by 60 foot wide semi-prepared runway and then return. Landing at a 2100 foot field would have required the use of powered-lift, stability and command augmentation, low approach speeds, and operation further on the backside of the power required curve than normally experienced by large transports. Safe and routine operation into and out of the 2100 foot field was required, which included control of an engine failure and completion of terminal area operations with the failed engine. Existing military requirements did not address all of these conditions and situations; therefore, it was necessary to prepare new requirements and modify existing ones for specific AMST application. The general specification for V/STOL Flying Qualities (MIL-F-83300, Reference 2) was not used extensively since the requirements were based mainly upon Vertical Takeoff and Landing (VTOL) flight.

The longitudinal requirements of MIL-F-8785B required the most modification. The short field Operational Flight Envelopes were modified to account for hot day operation (sea level/103°F, an AMST operational requirement), gust encounters and a minimum operational speed based on 1.2 times the stall speed. An AMST must be capable of encountering the same upgusts as encountered by conventional jet transports without stalling. These gusts were defined as 20 knots with all engines operating (AEO), and 15 knots with the critical engine inoperative (CEI), and will provide adequate angle of attack margin. Since a STOL airplane approaches at a lower airspeed than a similar conventional airplane, the angle of attack margin to stall must be larger than that normally provided. The gust magnitude was reduced with an engine failure because the probability of having an engine failure and encountering the higher gust magnitude was considered small.

The stall speed for the landing configuration was based on the following condition that results in the highest stall speed; either AEO at approach power on the engine(s), or CEI with takeoff power on the remaining engine(s). This stall speed would then be the reference used for all thrust settings. Adequate speed and horizontal gust encounter margins were provided by the 1.2 factor times this stall speed. In the event of an engine failure, the speed margin would provide the pilot with time to increase power from approach setting to takeoff power. As can be seen from the figure, the stall speed is a function of thrust setting.



Power settings significantly below that required for approach were experienced during initial intercept of a six degree glide slope from level flight. To insure a safe margin from stall when operating with low power settings, a stall warning system should be utilized.

Two further points are highlighted concerning desired control characteristics as the STOL configuration stall speed is approached. First, a requirement was added that prohibited the airplane from being flown at a speed lower than the minimum air control speed, V_{mc_a} (i.e., the minimum speed at which a dynamic engine failure could be safely controlled). This was different from the requirements of MIL-F-8785B which addresses V_{mc_a} only for the takeoff configuration. For conventional airplanes this is adequate since directional control with an engine failure is the primary concern. However, with the use of powered lift during a STOL landing the rolling moment due to an engine failure becomes significant and a V_{mc_a} concern can exist. Further, at the stall speed the lateral control must be sized to balance the static asymmetric rolling moments due to an engine failure. This requirement is compatible with the added V_{mc_a} requirement in that if one is met the other should be also.

Only two primary longitudinal controllers were permitted to control the airplane, for example, a stick and throttle. The requirement was based on the NASA Ames Research Center studies which

found that the pilot workload became unacceptably high if three controllers were used continuously during a STOL landing. A third controller might be used to command flap angle or thrust vector. A controller such as a direct lift switch on the throttle would be acceptable as long as the pilot did not have to remove his hands from the stick or throttles to operate it.

The flying qualities requirements for short landings were written such that either the normal backside control technique (i.e., column for airspeed and thrust for flight path) or the frontside control technique (vice versa) could be used. Use of the frontside control technique on the backside of the power required curve would require automatic commands to control surfaces and thrust to obtain the proper airplane response. Either control technique appeared to be acceptable for flight test pilots in terms of precision of control, training and safety. However, further study is necessary to determine if these results are also valid for low-flight-time pilots, and if pilots using a backside control technique would revert to a frontside control technique during an unexpected, hazardous situation.

For either control technique, it was desired to decouple the airplane response to pilot inputs to the greatest extent possible (i.e., make airspeed and flight path responses independent). For the frontside control technique, a requirement was added that the steady state attitude change to flight path angle change $\frac{\Delta\theta}{\Delta\gamma}$ fall within the following limits:

Level	$\Delta\theta/\Delta\gamma$	
	Min	Max
1	0.75	1.5
2	0.5	1.5

This requirement was based primarily on simulation results (Reference 3), and the intent was to make the STOL airplane respond as if it were operating on the frontside of the power required curve. That is, for a conventional airplane during a climb or descent at constant speed, the angle of attack is essentially constant. Hence, the following derivation indicated that a $\Delta\theta/\Delta\gamma$ of approximately one is required:

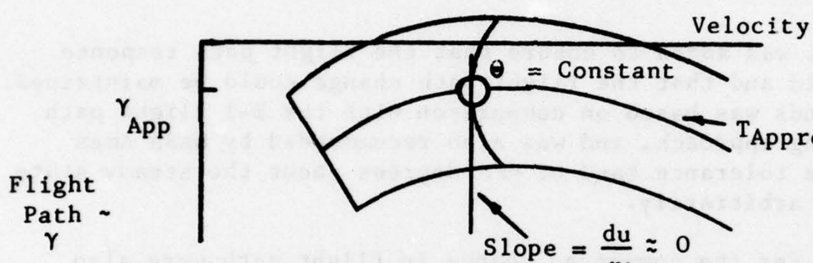
$\theta = \gamma + \alpha$

$\frac{\partial \theta}{\partial \gamma} = \frac{\partial \gamma}{\partial \gamma} + \frac{\partial \alpha}{\partial \gamma}$ and since $\frac{\partial \alpha}{\partial \gamma}$ is essentially zero for conventional response:

$\frac{\partial \theta}{\partial \gamma} = \frac{\partial \gamma}{\partial \gamma}$ or $\frac{\Delta \theta}{\Delta \gamma} = 1$

The requirement implies that automatic commands to control surfaces and thrust are provided in order to obtain the required $\Delta\theta/\Delta\gamma$ on the backside of the power required curve.

For the backside control technique, the requirement was that the slope of steady state velocity change to flight path angle change ($\Delta u/\Delta\gamma$) be approximately zero when pitch attitude is held constant. This is illustrated on the following figure:

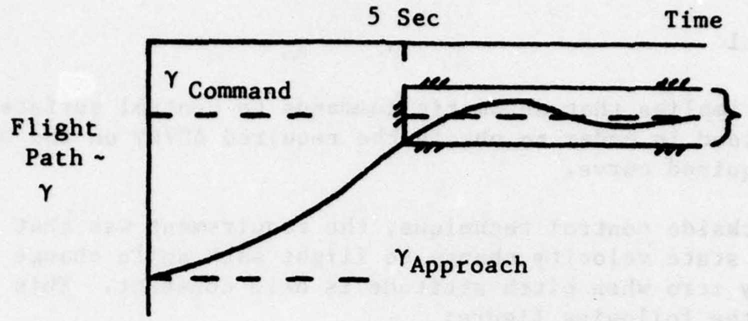


This requirement was based on discussions with NASA Ames. The intent was that thrust changes would result mainly in a flight path response, and not airspeed changes. Sufficient data were not available to specify a quantitative requirement.

Operation on the backside of the power required curve was restricted by limiting the positive values of the slope of flight path angle change to airspeed change $\frac{\Delta\gamma}{\Delta V}$ to values that did not restrict short field performance. NASA Ames had determined that operation on the backside of the power required curve was not a problem as long as sufficient thrust was available for flight path control. However, the USAF was concerned that sufficient thrust would not be available, so operation on the backside of the power required curve was limited by $\Delta\gamma/\Delta V$. The $\Delta\gamma/\Delta V$ requirement does not apply to an airplane that has a speed hold system since ΔV would be essentially zero and the slope then approaches infinity. The $d\gamma/dV$ values of the general MIL-F-8785B were retained for conventional takeoff and landing (i.e., at higher speeds and less flap deflection).

Requirements were added to those of MIL-F-8785B concerning the response of the flight path to a command, and the magnitude of the flight path change required for safe and routine STOL landings. The

flight path response requirement is illustrated below.



The requirement was added to ensure that the flight path response to command was rapid and that the flight path change could be maintained. The time of 5 seconds was based on comparison with the B-1 flight path response time during approach, and was also recommended by NASA Ames (Reference 4). The tolerance band of ± 0.5 degrees about the steady state value was selected arbitrarily.

Various values for the commanded change in flight path were also defined. An AMST with all engines operating must have the capability to obtain from the initial STOL flight path both level flight or a negative flight path change of 4 degrees. An AMST that sustains an engine failure below the go-around decision height must reach the runway and land in winds up to a 16 knot headwind or 6 knot tailwind. When a landing is made after an engine failure has occurred above the go-around decision height, an AMST must be able to change flight path angle by ± 2 degrees in clam air, as well as be able to compensate for 23 knot headwind or 10 knot tailwind.

These requirements were added to specify the minimum flight path control required for safe and routine STOL operation. The requirement may have a direct impact on sizing the engines since thrust would be the primary means of controlling the flight path on the backside of the power required curve. For this reason, this requirement was discussed more than all the other changes made to the general MIL-F-8785B and was the most controversial. The basis for the requirement was simulation and flight test studies done by the NASA Ames Research Center (References 4 and 5). Capability to change the flight path was necessary to track to a desired touchdown point and to correct flight path errors due to the pilot, turbulence or wind shears. Three different cases were addressed in the requirement:

All Engines Operating (AEO), and Critical Engine Inoperative (CEI) below and above the engine-out go-around height. The critical case would be the CEI landing where the engine failure occurred above the go-around decision height. Maximum available thrust would be reduced due to the engine failure, and since the probability of this case occurring would be greater than the one where an engine failure occurred below the go-around decision height, higher wind shear magnitudes must be accounted for in the design. The magnitudes defined in the requirement were based on ASD judgment. No manually commanded configuration changes were allowed once the landing approach was begun in order to reduce pilot workload. Further, the configuration change required to optimize flight path change capability would be difficult to determine since it would be a function of the winds or shears encountered. The configuration change would also take time which may not be available when close to the ground. Manual configuration changes would take longer than automatic since, in general, command would be stated verbally by the pilot to the copilot who must then react.

The flight path change of -4 degrees with all engines operating would result in a total flight path angle of -10 degrees for a -6 degrees STOL approach angle. A capability near this magnitude was required for capturing the STOL glide slope from level flight. Negative corrections to the flight path are not as critical as positive corrections since if the pilot cannot correct downward, a go-around could be commanded and safety not compromised. The pilot must make upward corrections or risk short or high sink rate landings. It is strongly recommended that future research be conducted to determine the flight path change capability required for safe STOL landings, particularly because of the significant impact the requirement can have on both engine size and cost.

The frequency and damping requirements of the general MIL-F-8785B were modified to account for STOL unaugmented mode characteristics, as well as for augmented mode characteristics with all systems operational. The lower bounds on the unaugmented short period frequency requirement (Level 3) for the landing flight phase were relaxed if the airplane complied with the flight path dynamic response requirement discussed above. The reason for this was that the frequency requirement was intended to provide a minimum level of load factor or flight path control through the longitudinal controller. Since the thrust is used to control flight path for a STOL airplane it appeared reasonable to relax the frequency requirement if the flight path response due to thrust was acceptable.

The longitudinal damping requirements are shown in Figure 1. The Levels 1, 2, and 3 requirements apply for all categories, and the solid lines on Figure 1 were based on the general MIL-F-8785B (Reference 1). The damped frequency of 0.3 rad/sec was selected based on AMST root location as a reasonable boundary between short

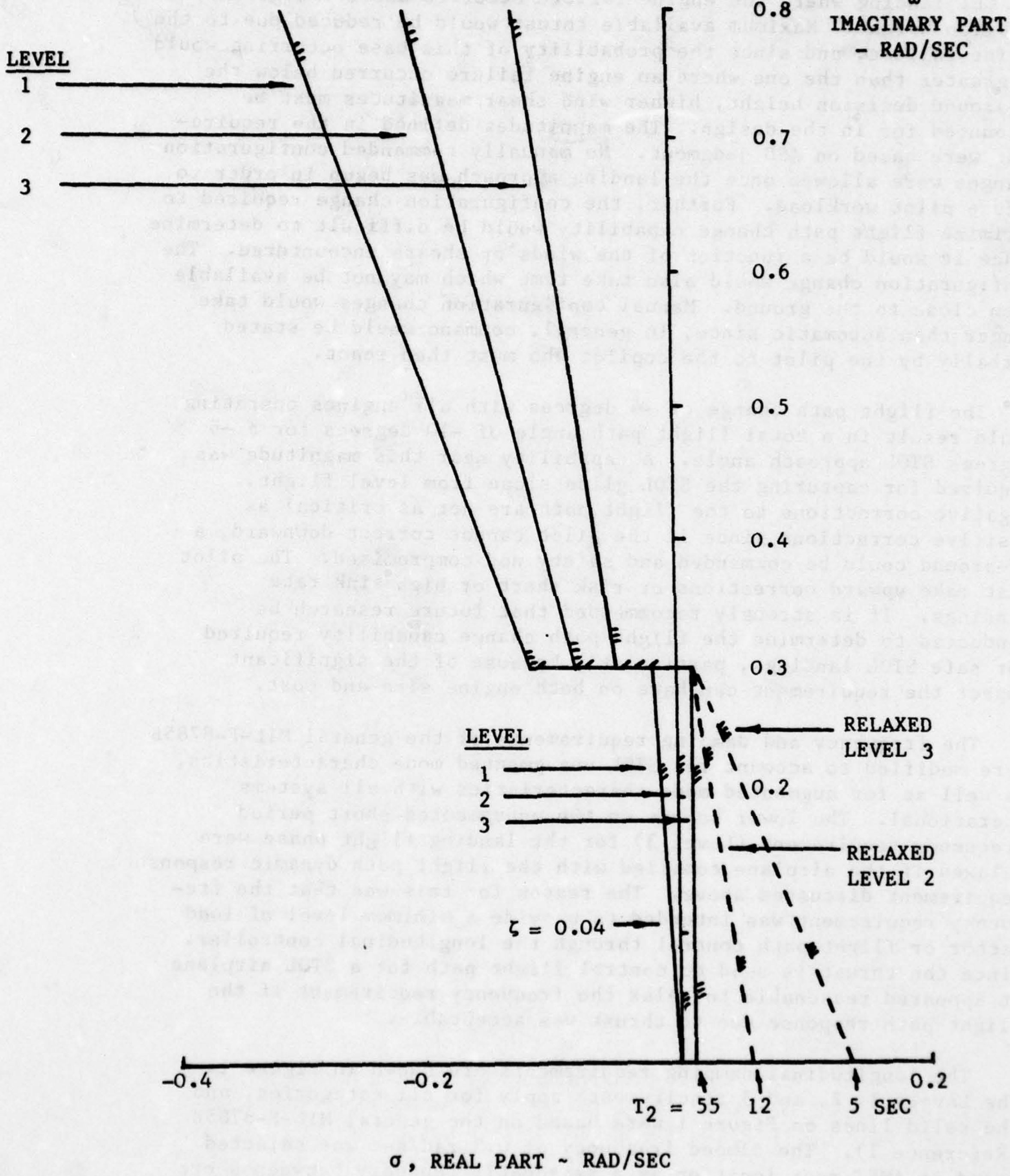


FIGURE 1. SYSTEM DAMPING

duration and long duration modes. The relaxed regions for Levels 2 and 3 (dashed lines) were based on SST approach simulation study results (References 6 and 7) which indicated safe flying qualities were possible with root locations even more unstable than allowed by Figure 1. Roots may be in the relaxed areas for only short periods of time to prevent high pilot workload or fatigue. These damping requirements were considered adequate for the AMST program since both prototypes had damonstrated good damping characteristics. Further study is required to develop a general longitudinal damping requirement that addresses all system modes in a systematic and nonsubjective fashion.

In general, the lateral-directional requirements of the general MIL-F-8785B were used for the AMST. Several notable exceptions are discussed below. The AMST Level 1 roll control capability for Category A tasks was relaxed from the general MIL-F-8785B heavy airplane (Class III) requirement of 30 degrees bank angle in 1.5 seconds to 30 degrees in 2.0 seconds. The AMST Level 1 requirement was based on AMST demonstrated capability, which was found to be acceptable during STOL operation with an engine failed, MAC tactical maneuvers, in turbulence, crosswinds and in formation flight. The roll requirements must be met at the minimum operational speed of 1.4 V_s for Categories A and B (aerial delivery excluded), and 1.2 V_s for Category C (including aerial delivery).

A capability to land in a crosswind of 30 knots, measured at 50 feet altitude, was required at the short field approach speeds. Specifying the altitude at which the crosswind was measured was necessary since the new turbulence model in the AMST requirements accounted for wind shear. A crosswind of 30 knots at the STOL landing speeds resulted in a sideslip angle near 20 degrees. This may be the critical design case for rudder sizing, and selection of the vertical tail airfoil to avoid fin stall.

The minimum ground control speed, V_{mc_g}, the speed during takeoff at which an engine failure can occur and the pilot can keep the airplane on the runway during a continued takeoff, must be based on the narrow STOL field width of 60 feet. This meant that lateral deviations from the runway centerline had to be less than 20 feet to keep the main landing gear on the runway. This was a necessary requirement for safe operation, but it could significantly affect tail size, hydraulic sizing for high rudder rates or flight control yaw damper logic depending upon how fast a pilot can be expected to react. Since the pilot reaction time is an integral part in determining V_{mc_g}, the airplane design to yield low V_{mc_g} values can become quite subjective since pilot reaction time can vary from zero to several seconds, depending upon pilot anticipation and cues.

As mentioned above, the atmospheric model for the AMST specification was different from the general MIL-F-8785B. STOL airplanes flying on the backside of the power required curve and lower approach speeds are more susceptible to changes in flight path, angle of attack and airspeed when gusts and shears are encountered. Thus, it was necessary to model the gusts and shear characteristics accurately to determine actual STOL capability. The AMST model was a simplified version of the model developed for the Federal Aviation Administration for flight certification of airplanes by simulation (Reference 8).

In conclusion, this paper has addressed some of the changes made to the general MIL-F-8785B for the AMST. A more complete discussion of this topic is presented in "USAF Flying Qualities Requirements for a STOL Transport" (ASD-TR-78-13).

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Dwight Schaeffer, Boeing: Why are the $\partial\gamma/\partial V$ required for CTOL much more stringent than the STOL requirements?

Answer: The $\partial\gamma/\partial V$ requirements for CTOL are more stringent since pilots trained only in CTOL flight requiring front side of the power required curve piloting technique must be restricted from operation very far back on the power required curve so that the front side technique still works well.

Frank Wilson, Lockheed-Georgia: C-130 STOL flight simulation in the presence of wind shear showed that powered lift aircraft tend to be much more tolerant of wind shear than do others such as the C-141 or C-5. Reason is that power advance for go-around provides DLC simultaneously with longitudinal acceleration which greatly minimizes altitude loss.

Answer: In general, it is felt that STOL airplanes are more susceptible to changes in angle of attack, airspeed and altitude when shears are encountered than conventional airplanes. Results of C-130 STOL flight simulation are directly related to shears encountered and speed of approach.

IMPACT OF KEY FLYING QUALITIES REQUIREMENTS ON THE YC-14/C-14 DESIGN

E. Frank Carlson

The Boeing Company

SUMMARY

Some of the flying qualities requirements that have a major impact on the YC-14/C-14 design are discussed. Comments are made on selected portions of the proposed revisions to MIL-F-8785B, and some suggestions for future revisions to the flying qualities specification (or standard) are presented. Particular emphasis is placed on the low speed flight regime, and on the stability and control augementation system. For instance, the definitions of stall speed and some minimum operating speeds should be modified to account for characteristics associated with a highly augmented, powered lift aircraft. The proposed revision to the definition of flying qualities levels is discussed with emphasis on the interface between flying qualities levels, atmospheric disturbances, and failure states. Also, some thoughts are presented on the application of certain requirements to an augmented aircraft.

DISCUSSION

Before discussing the individual flying qualities requirements, it would by well to first consider the general philosophy to be applied to today's more highly augmented aircraft.

A highly augmented airplane, such as the YC-14, presents a peculiar set of problems when applying the MIL-F-8785B requirements. For instance, the YC-14 flight control system provided the pilot with rate command (stick out of detent) and attitude hold (stick in detent). This is the normal "manual" mode of operation, not a pilot assist mode. The requirements in MIL-F-8785B are directed toward a rate command system. Trying to impose some of these requirements on an attitude hold system can compromise the system design and degrade its performance. An example is found in MIL-F-8785B section 3.2.2.1.1 (short period frequency and acceleration sensitivity). It is doubtful that an attitude hold type system was considered when formulating this requirement. The designer must go to MIL-F-9490D to find requirements for an attitude hold system. Use of MIL-F-9490D does not provide a complete solution since it gives system performance requirements, but does not specify flying qualities requirements per se. A similiar situation exists for a speed control system that is a part of the normal manual control mode of operation.

To alleviate problems with these various manual control modes it is recommended that: a) the applicable control mode(s) (rate command, attitude hold, etc.) should be identified for each section of MIL-F-8785B, where such identification is appropriate, and b) additional sections be incorporated into MIL-F-8785B to define those requirements not already covered adequately in MIL-F-8785B for the additional manual control modes such as attitude hold and airspeed select/hold.

The following discussion of the various flying qualities topics is organized in the same sequence as the subject headings in MIL-F-8785B; thus,

the various subjects are NOT arranged by their relative order of importance.

1.5 LEVELS OF FLYING QUALITIES

Recognition of the effects of atmospheric disturbances on Flying Qualities Levels, as presented in Reference 1, is a step in the right direction. However, the particular definition of Levels proposed for revising MIL-F-8785B (Reference 1) will lead to confusion and misunderstandings, particularly during flight testing. The pilot would have to make an assessment of the atmospheric disturbance intensity before he could rate the Flying Qualities characteristics as a particular Flying Qualities Level. This would introduce significant additional variability in pilot ratings assigned by different pilots.

The atmospheric disturbance severity should not be an integral part of the definition of Flying Qualities Levels since it is more appropriately considered as part of the task or the task environment. An alternate approach to recognizing the effects of atmospheric disturbances on Flying Qualities Levels is as follows:

- o Retain the current definition of Flying Qualities Levels as given in MIL-F-8785B, Reference 2
- o Integrate the atmospheric disturbance requirements into section 3.1.10, "Applications of Levels". Suggestions for this are given below in the discussion of section 3.1.10.

This will maintain the required separation of the pilots assessment of Flying Qualities Levels and the task(s) he must perform.

A second area of concern in the definition of Flying Qualities Levels is the relationship between the definitions of Levels in MIL-F-8785B (Reference 2) and the definitions used in the Cooper-Harper pilot rating scale. Specifically, Level 3 is defined in MIL-F-8785B as:

Level 3 Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.

It is assumed that this definition means the Category B and C Flight Phases must be completed successfully. Successful completion of these Flight Phases (particularly landing) is interpreted as requiring a Cooper-Harper pilot rating better than 7, since a rating of 7 is defined as "Adequate performance not attainable with maximum tolerable pilot compensation..." (misses the runway and/or damages the airplane). Note, however, that section 1.5 of the MIL-F-8785B users guide (Reference 4) equates a Cooper-Harper pilot rating of 6.5 or better with Level 2. Thus, an airplane that is Level 3 under the MIL-F-8785B definition could be given a Cooper-Harper pilot rating of 6.5, which is equivalent to Level 2 when using the relationships given in the users guide (Reference 4). I believe that some discussion is needed to resolve this inconsistency between the two pilot rating systems.

3.1.5 CONFIGURATIONS

The current definition of configurations in Reference 2 can pose some interesting problems for the designer of an augmented aircraft. Take, for example, an aircraft on landing approach that only meets the Level 2 requirements because it operates on the back side of the power required curve. Now suppose that this aircraft uses a redundant speed hold system to provide Level 1 flying qualities for the landing approach task. A literal interpretation of the configuration definition requirements in MIL-F-8785B (Reference 2) would require that if the pilot were to be given direct control over engaging/disengaging the speed hold system, then the aircraft would have to meet the Level 1 requirements with the speed hold system disengaged as well as with the system engaged. Thus, to satisfy this requirement, it would be necessary to have the speed hold system automatically engage when landing flaps are selected. Further, no pilot override or provision to switch the redundant speed hold system off would be permitted since the aircraft would then be required to meet the Level 1 requirements with the speed hold system both engaged and disengaged. Neither of these would be a satisfactory situation since the pilot should have direct control over the time when the speed hold system is engaged/disengaged. The preceeding logic can be extended to include redundant stability and/or control augmentation systems.

It is recommended that the definition of configurations in MIL-F-8785B, Reference 2, be reworded to make provisions for redundant augmentation systems.

3.1.7 OPERATIONAL FLIGHT ENVELOPES

Two subjects will be considered in this section. The first deals with the potential benefits of augmentation systems on the Flight Phase Category C minimum operating speeds. The second subject deals with the effects of powered lift on the Flight Phase Category A minimum operating speeds.

Typical design margins for a transport aircraft call for the Flight Phase Category C minimum operating speed to be 20 percent above the "stall" speed, i.e., $V_{o_{min}} = 1.2\ V_s$. Although it was not possible to make a full evaluation during the limited YC-14 flight test program, the pilots felt that the YC-14 could be flown safely and routinely with a reduced speed margin, such as $V_{o_{min}} = 1.1\ V_s$. This confidence resulted from a variety of factors, primarily:

1) The speed hold system maintained speed accurately, thereby preventing the pilot from inadvertently slowing down.

2) The stability and control augmentation system provided rate command (controller out of detent) and attitude hold (controller in detent). The attitude hold feature reduced the pilot workload.

3) Flight control system redundancy, including the speed hold, provided fail operational capability after the first failure and fail safe operation following a second failure.

4) Engine out protection was provided through automatic sensing and subsequent reconfiguration of the flaps optimized performance and minimized the flight path perturbations following an engine failure.

5) The pilot workload was significantly reduced as a result of the flight control system features described above.

6) Rapid engine response, as a result of the reasonably high approach power settings, enabled the pilot to make rapid speed changes if the need arose.

Without the control system features incorporated in the YC-14, the pilot could encounter difficulties at lower than normal approach speeds such as 1.1 V_S; thus, system reliability requirements became a significant factor in determining the minimum normal approach speeds. These reliability requirements can be handled in an orderly manner using the approach outlined in section 3.1.10.

It is recommended that the flight control system features and capabilities be considered carefully in interpreting the current MIL-F-8785B minimum operating speed requirement which is "minimum normal approach speed" for landing approach. With proper flight control system design it would seem reasonable to reduce many of the Flight Phase Category C minimum operational speed margins below the traditional levels while maintaining adequate wind shear/gust protection and turn rate capability.

The second subject to be considered in this section deals with Flight Phase Category A minimum operating speeds. Powered lift airplanes such as the YC-14/C-14 tend to have relatively low "stall" speeds; thus, the minimum operating speeds, which are some percentage above the "stall" speeds, are also relatively low. These low minimum operating speeds can impose some stringent design requirements on the aircraft. This is particularly true for the Flight Phase Category A roll performance requirements in section 3.3.4 of MIL-F-8785B.

The need for the relatively low minimum operational speed capability for some Category A Flight Phases should be considered further. For instance, the in-flight refuel (receiver) requirement would provide a minimum operating speed capability well below that for any available or projected tanker airplane. For example, the minimum operational speed for the YC-14 as a receiver is 150 kts (wt = 140,000 lb) while a typical minimum operating speed for refueling with the KC-135 tanker is 183 kts (wt = 170,000 lb). Typical refuel speeds for the KC-135 range from 250 to 320 kts.

As currently stated, the MIL-F-8785B rules would require the YC-14/C-14 to have significant capabilities to meet the Flight Phase Category A requirements at speeds well below those required in service. This additional capability can add significantly to the aircraft cost. It is recommended that the minimum operating speeds be specified as the higher of 1) some fraction of V_S as is now done, or 2) finite speeds which are compatible with the needs of the Flight Phase.

AD-A066 493 AIR FORCE FLIGHT DYNAMICS LAB WRIGHT-PATTERSON AFB OHIO F/G 1/2
PROCEEDINGS OF AFFDL FLYING QUALITIES SYMPOSIUM HELD AT WRIGHT --ETC(U)
DEC 78 G T BLACK, D J MOORHOUSE, R J WOODCOCK

UNCLASSIFIED AFFDL-TR-78-171 NL

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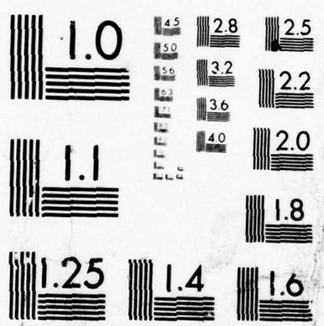

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

3.1.10 APPLICATIONS OF LEVELS

As mentioned previously in section 1.5, the effects of atmospheric disturbances should be considered in this section, along with Airplane Normal States and Failure States. Two variants to this integrated approach are shown in Figures 1 and 2, where both figures present essentially the same requirements. The first figure is based on a smooth, continuous change with failure probability while Figure 2 is based on discrete values. The data shown in these figures pertain to a Class III airplane. The values of some of the numbers would be different for other airplane classifications. The proposed approaches shown in Figures 1 and 2 have the following advantages:

1) Permits Flying Qualities Levels to be defined independent of the environment.

2) Considers that the probability of encountering both severe atmospheric disturbances and a hazardous failure simultaneously is remote. This is not to say that it is impossible, but only that it becomes sufficiently unlikely that it is not mandatory to compromise the airplane design to cover these conditions.

3) Considers the probability of the airplane being in or of entering into the more extreme Flight Envelopes (Service and Permissible).

4) The Airplane Normal State and Failure State requirements in the figures are compatible with the corresponding sections 3.1.10.1 and 3.1.10.2 of MIL-F-8785B. Figures 1 and 2 are also compatible with the turbulence and safety sections (3.1.3.7.1 and 3.1.7) of MIL-F-9490D, Reference 3.

5) Figure 2 shows a lower wind/turbulence probability of exceedence for Flight Phase Category C than for Categories A and B. The probability of exceedence was reduced for Category C since the exposure time is lower than in Categories A and B. The same type of distinction could be specified under the format used in Figure 1.

The format in Figure 1 appears more desirable since it permits the designer to more easily visualize the requirements and select the critical design points.

3.2.1.1 LONGITUDINAL STATIC STABILITY

The proposed revision to this section of MIL-F-8785B would permit a single unstable mode of motion with a time to double amplitude of no less than 6 seconds for Level 3 operation. Results from Boeing and other programs indicate that this level of instability should be acceptable. The Concord SST was certified with a 5 second time to double amplitude pitch instability for the landing approach task with the stability augmentation system inoperative. The TIFS was used to evaluate the Concord at this critical flight condition. The conclusions, given in Reference 5, state "the minimum acceptable boundary (Pilot Rating of 6.5 on the Cooper-Harper scale) determined in this investigation occurs at a T_2=2.5 seconds in light turbulence ($\sigma_{v_{g_{max}}}$ = 1.5 ft/sec) and at a T_2=4.25 seconds for

moderate turbulence ($\sigma_{V_{g_{max}}}$ = 3.0 ft/sec)". The SST being developed by Boeing in the late 60's and early 70's also exhibited an unstable pitch mode of motion on landing approach with augmentation system failures. Extensive piloted simulations demonstrated that a time to double amplitude of 6 seconds met Boeing requirements for Cooper-Harper pilot ratings better than 6.5. Thus, it is concluded that the proposed Level 3 limit of 6 seconds time to doublé amplitude is conservative, and could safely be reduced to less than 5 seconds for a single unstable mode of motion.

Another factor to consider is spiral mode stability. The current version of MIL-F-8785B permits an unstable spiral mode for all Flying Qualities Levels. The proposed revision to section 3.2.1.1 would permit only a single unstable mode of motion; thus, if there is an unstable pitch mode, the commonly accepted spiral instability would be precluded. It is recommended that the proposed revision to MIL-F-8785B be revised to allow an unstable spiral mode in combination with a single unstable longitudinal mode of motion.

3.2.2.1.1 SHORT-PERIOD FREQUENCY AND ACCELERATION SENSITIVITY

The Level 3 frequency requirement given in this section of the basic MIL-F-8785B should be reviewed in light of the proposed revision to section 3.2.1.1. Typically, as an unaugmented airplane becomes less stable the short period roots will change from an oscillatory pair to a pair of real roots, see Figure 3. The frequency of the dominant "short period" root decreases as the stability is further reduced. Usually this dominant "short-period" root will couple with one of the "phugoid" roots, forming an oscillatory pair. This frequently occurs as the unstable real root is on the order of a time of double of 6 seconds. These characteristics, allowed under the proposed revision to section 3.2.1.1, are in conflict with the minimum Level 3 frequency requirements of 3.2.2.1.1 of MIL-F-8785B.

It is recommended that the short-period frequency requirement for Level 3 be waived when an unstable pitch axis root is permitted under section 3.2.1.1.

3.2.2.2.1 CONTROL FORCES IN MANEUVERING FLIGHT

In the proposed revision to this section, item number 2 in the "rationale for revision" states the case well for providing structural/stall protection via a nonlinear increase in stick force near n_L and/or stall. This desired type of protection could, however, be precluded by the specific wording of the proposed revision to this section. Take, for example, a Class III airplane in one of the more demanding flight phases in the Operational Flight Envelope where $n_{o_{MAX}} = n_L = 3$. The F_s/n_z must be approximately linear up to $n_z = .5\left[n_{o_{MAX}} + 1\right] = 2$. Above $n_z = 2$ an increase in F_s/n_z of "more than 50 percent is considered excessive". An increase in F_s/n_z of only 50% between $n_z = 2$ and 3 will not, in many cases, provide the necessary degree of structural/stall protection. Therefore, it is recommended that the 50% linearity requirement be modified for Class I, II and III airplanes. The 50% linearity requirement should be imposed only on reductions in the control force gradient. Restrictions should be removed, or at least opened up, on the control force linearity at load factors greater than $.5\left[n_o(+) + 1\right]$.

3.3.4.2 ROLL PERFORMANCE FOR CLASS III AIRPLANES

The proposed revision to the Class III airplane roll performance requirements correlates well with results from the YC-14 program.

3.5.5.1 FAILURE TRANSIENTS

This proposed revision to the allowable failure transient magnitudes for Level 1 and 2 operation would appear to impose unduly restrictive requirements. The vertical or lateral excursion limits of 5 feet and the ±2 degrees of bank angle limit appear tight for Flight Phase Category A tasks, and even more stringent for Category C tasks. For Category B tasks it would seem reasonable to relax the requirements significantly.

6.2.2 SPEEDS

Before getting into the detailed definition of "stall" speed, V_S, it would be well to comment on the tendency to categorize airplane operation as either STOL (short takeoff and landing) or CTOL (conventional takeoff and landing). One of the first problems in distinguishing between STOL and CTOL operations is in the definition of terms. "STOL" has vastly different meanings to different people. A survey of personnel associated with the AMST program resulted in fourteen major classifications under which they defined "STOL", and further variations of the definitions within each of these major classifications. For example, some defined "STOL" in terms of field length, somein terms of approach speed, and some in terms of the rules used to define V_S. The type of operation described by some people as "STOL" would be better classified as "assault" operation since they are considering flight conditions where a failure, such as loss of an engine, would result in loss of the airplane.

It is felt that trying to distinguish between STOL and CTOL is, in general, unwarranted for flying qualities, and it can lead to problems and confusion in the definition of minimum speeds. It is proposed that a common set of rules could be applied to both the so called STOL and CTOL operations. This proposed commonality would be particularly valid for the YC-14 since the flight control system was designed to enable the pilot to use the same piloting techniques in all flight phases from high speed down through landing on a 2000 foot field.

The definition of the "stall" speed, V_S, can have a profound influence on the design of a powered lift aircraft. This was particularly true for the YC-14 which used a speed control system to command both throttle and USB (upper surface blowing) flap position to maintain speed without disturbing the airplane's flight path. The YC-14 system mechanization was such that when the pilot dialed in a higher flight speed, the thrust would increase and the USB flap would retract enough to maintain approximately constant lift at the new speed. This movement of the USB flap redirected the engine thrust in a manner to increase the powered lift effects as the speed decreased; thus, making it possible to fly safely at speeds lower than would be possible with a fixed flap. The fail operational/fail passive and various saturation protection features built into the YC-14 speed control system helped to assure flight safety.

Several factors should be considered in the definition of V_s to take full advantage of powered lift while assuring flight safety. The important considerations are:

1) Factors such as excessive rate of sink that are not now considered in the definition of V_s.

2) Thrust management and the use of a speed hold system that may include flap configuration management. The manner in which such a system is used in determining V_s can become significant, as can the failure modes and redundancy level.

3) The flight safety implications of an engine failure, both in terms of controllability and go-around climb performance capability.

V_s as defined in the AMST proposal instruction package included the factors discussed above. This definition is shown in Figure 4. It appears that this definition of V_s is equally applicable to other airplane types, thus it is recommended for adoption in MIL-F-8785B.

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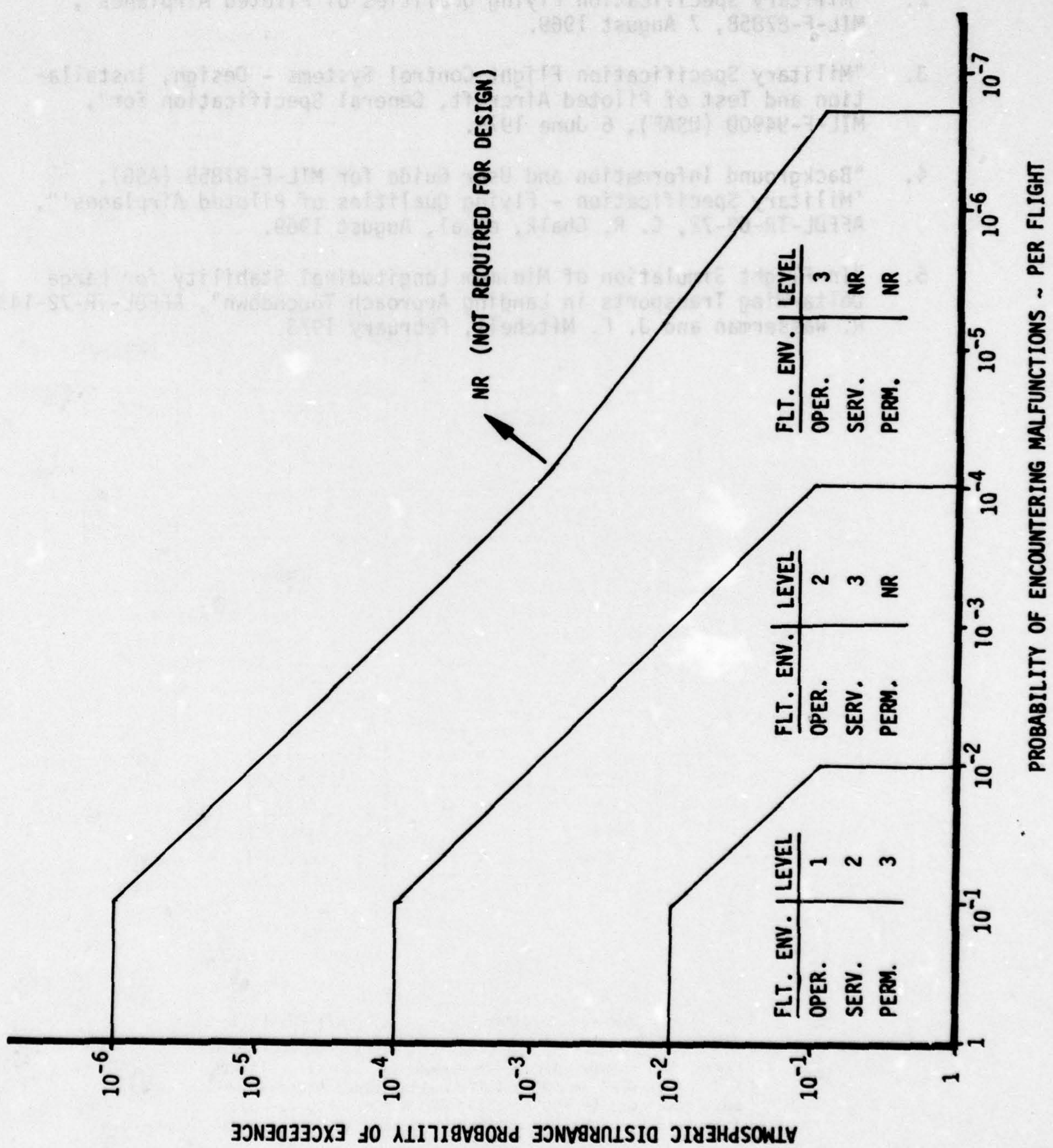

FIGURE 1 APPLICATION OF LEVELS

FIGURE 2b

FLYING QUALITIES REQUIREMENTS - GENERAL FAILURE STATES - PEACETIME CONDITIONS

- "FLIGHT CRITICAL SYSTEMS" FAILURES
- "NORMAL" OPERATING MODE
- FLYING QUALITIES MAY EXCEED MINIMUM LEVELS SHOWN

PROBABILITY OF ENCOUNTERING MALFUNCTION(S) ~ PER FLIGHT	MINIMUM FLYING QUALITIES LEVEL AFTER MALFUNCTION		
	OPERATIONAL	SERVICE	PERMISSIBLE
Greater Than 10^{-2}	Level 1	Level 2	Level 3
Less Than 10^{-2}	Level 2	Level 3	NR [1]
Less Than 10^{-4}	Level 3	NR	NR
Less Than $5x10^{-7}$	NR [1]	NR	NR

The above flying qualities levels must be maintained up to wind/turbulence severity associated with the following probability of exceedence

PROBABILITY OF ENCOUNTERING MALFUNCTION(S) ~ PER FLIGHT	WIND/TURBULENCE PROBABILITY OF EXCEEDENCE					
	OPERATIONAL		SERVICE		PERMISSIBLE	
	CATEGORIES A & B	CATEGORY C	CATEGORIES A & B	CATEGORY C	CATEGORIES A & B	CATEGORY C
Greater Than 10^{-2}	10^{-2}	10^{-1} [2]	10^{-2}	10^{-1} [2]	10^{-2}	10^{-1} [2]
Less Than 10^{-2}	10^{-2}	10^{-1}	10^{-1}	$10^{-0.5}$	NR [1]	NR
Less Than 10^{-4}	10^{-1}	$10^{-0.5}$	NR	NR	NR	NR
Less Than $5x10^{-7}$	NR	NR	NR	NR	NR	NR

In addition, for Category C in the Operational Flight Envelope, the flying qualities shall meet or exceed:

PROBABILITY OF ENCOUNTERING MALFUNCTION(S) ~ PER FLIGHT	WIND/TURBULENCE LEVEL PROBABILITY OF EXCEEDENCE [3]		
	10^{-1}	10^{-3}	10^{-5}
Greater Than 10^{-2}	Level 1	Level 2	Level 3
Less Than 10^{-2}	Level 2	Level 3	NR
Less Than 10^{-4}	Level 3	NR [1]	NR
Less Than $5x10^{-7}$	NR	NR	NR

[1] "NR" means Not Required for design - aircraft may be lost due to malfunction(s).

[2] Category C crosswind/tailwind component limits at 13 KTS/ 6 KTS (measured at 50 ft.) for malfunctions occurring no more than once in 100 flights, and 8 KTS/4 KTS for malfunctions occurring no more than once in 10,000 flights.

[3] The crosswind/tailwind limits from Fig 2a shall apply.

FIGURE 2a

MINIMUM REQUIRED FLYING QUALITIES FOR

- AIRCRAFT NORMAL STATES
- "NORMAL" OPERATING MODE
- VARYING WIND/TURBULENCE SEVERITY

HANDLING QUALITIES REQUIRED	WIND/TURBULENCE PROBABILITY OF EXCEEDENCE [1]					
	OPERATIONAL FLT ENVELOPE		SERVICE FLT ENVELOPE		PERMISSIBLE FLT ENVELOPE	
	CAT A & B	CAT C	CAT A & B	CAT C	CAT A & B	CAT C
Level 1	10^{-2}	10^{-1}	NR	NR [2]	NR	NR
Level 2	10^{-4}	10^{-3}	10^{-2}	10^{-1}	NR	NR
Level 3	10^{-6}	[3] 10^{-5}	10^{-4}	10^{-3}	10^{-2}	10^{-1}

[1] Flying qualities level must be maintained up to wind and turbulence severity associated with specified probability of exceedence

[2] "NR" means Not Required for design.

[3] Category C CROSSWIND/TAILWIND COMPONENT LIMITS:

FLYING QUALITIES (OPERATIONAL ENVELOPE)	90° CROSSWIND AT 50 FT.	180° TAILWIND AT 50 FT.
Level 1	13 KTS	6 KTS
Level 2	30 KTS	10 KTS
Level 3	30 KTS	15 KTS

- Landing flying qualities may be satisfied by increasing indicated airspeed beyond V_{TH} by an amount equal to one-half the mean wind plus the "peak" gusts, not to exceed an increase of 20 knots.
- For Category C turbulence roughness length of Z = .15 ft. shall be used for design

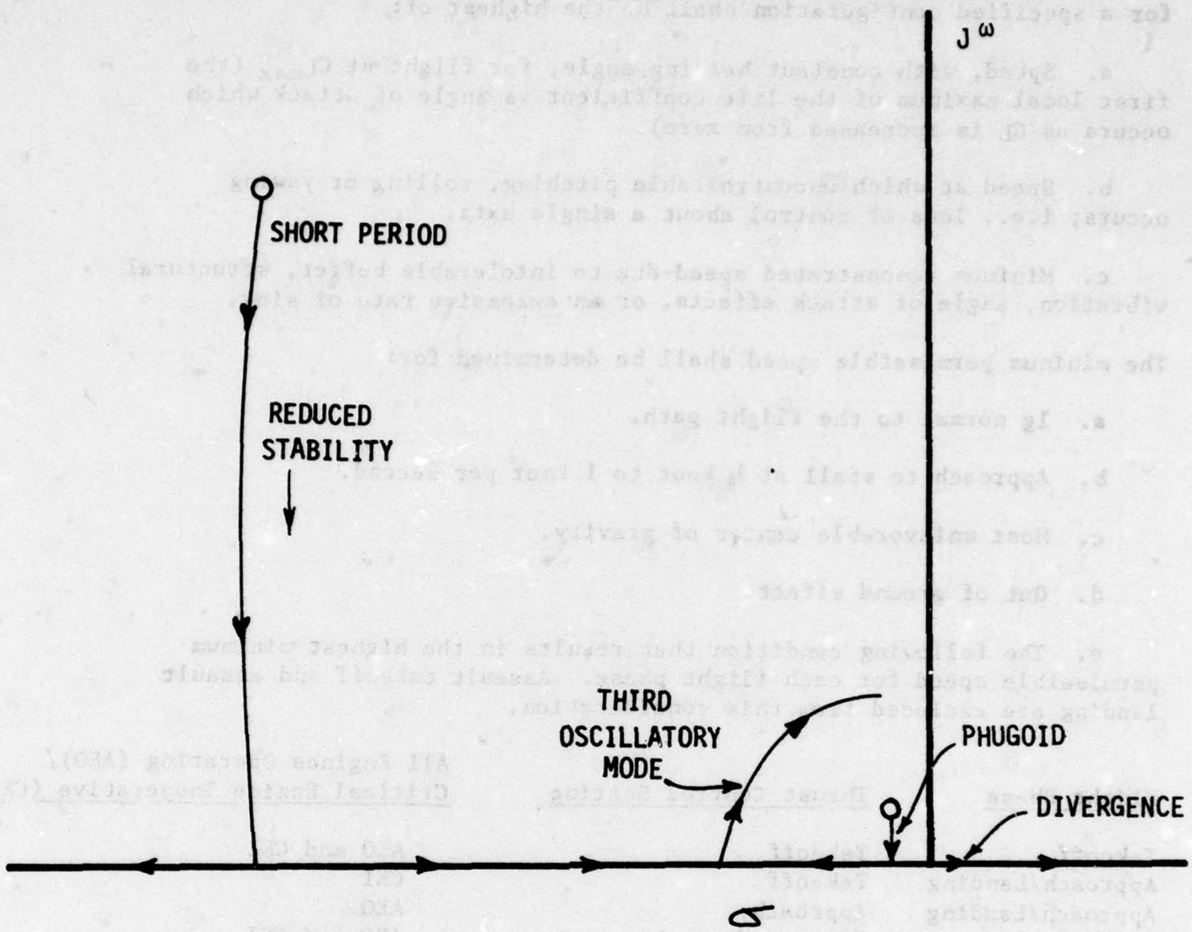

FIGURE 3 LOCUS OF PITCH AXES ROOTS AS CENTER OF GRAVITY MOVES AFT

3.1.9.2 Minimum Permissible Speed (V_s). The minimum permissible speed for a specified configuration shall be the highest of:

a. Speed, with constant heading angle, for flight at $C_{L_{max}}$ (the first local maximum of the lift coefficient vs angle of attack which occurs as C_L is increased from zero).

b. Speed at which uncontrollable pitching, rolling or yawing occurs; i.e., loss of control about a single axis.

c. Minimum demonstrated speed-due to intolerable buffet, structural vibration, angle of attack effects, or an excessive rate of sink.

The minimum permissible speed shall be determined for:

a. 1g normal to the flight path.

b. Approach to stall at ½ knot to 1 knot per second.

c. Most unfavorable center of gravity.

d. Out of ground effect.

e. The following condition that results in the highest minimum permissible speed for each flight phase. Assault takeoff and assault landing are excluded from this consideration.

Flight Phase	Thrust Control Setting	All Engines Operating (AEO)/ Critical Engine Inoperative (CEI)
Takeoff	Takeoff	AEO and CEI
Approach/Landing	Takeoff	CEI
Approach/Landing	Approach	AEO
Go-Around	Go-Around	AEO and CEI
Descent/ Deceleration	Minimum Allowable*	AEO and CEI
All Other	TLF at 1.2 V_s	AEO and CEI

*Minimum allowable includes reverse thrust.

FIGURE 4 "Stall" speed definition from the AMST proposal instruction package.

Jerry Lockenour, Northrop: Were simulations conducted by Boeing in setting (or verifying) the 6 sec. T_2 on SST, etc.? Did these simulations include a good simulation of flare and touchdown?
Answer: Yes, simulations were conducted. Yes, the touchdown simulation included the best simulation possible for flare and touchdown.

Bill Rickard, DAC: T_2=6 sec. is unconservative for moderate to heavy turbulence, or for long term tasks. DAC opposes this revision.
Answer: The revision to the spec does not even call for Level 3 type failure probabilities (equated with encountering "Level 3") in moderate to heavy turbulence. Thus, the question has no bearing on the discussion. I agree that the T_2= 6 sec. is only acceptable over short time periods such as the 5 min. allowed in the AMST spec.

Chick Chalk, Calspan: Slow divergence is not a factor in closed loop operation.
Answer: I agree.

Bob Woodcock, AFFDL/FGC: I'm confused about what leads you to think that we want to require the same flying qualities, augmentation on and off?
Answer: No, I think I have interpreted this section of the spec correctly. The YC-14 speed command/hold system could be switched on or off at will by the pilot. Thus, it could be considered "normal operation" in either case. The problem comes in landing where the speed command/hold system is required to achieve Level 1. It becomes immediately apparent to the pilot at speeds used for landings on a 2000 ft field. Thus, it should not be required to meet the requirements in succeeding sections required under the current sording in section 3.1.5.

Similar problems could occur with a stability augmentation system where a normal operation could consist of switching the augmentation system off for training. Since this then becomes

"normal operation" it could be construed to require Level 1 with the system off. These topics are explained in greater detail in my paper.

Tim Sweeney, ASD: I still believe that you have misinterpreted that paragraph of the specification. It does not say that Level 1 requirements have to be met with selectable configurations engaged and disengaged. It specifically states that all configurations REQUIRED or ENCOUNTERED in the flight phase are covered. This wording catches situations such as are encountered on aircraft with a single-channel SAS if the pilot is instructed to turn it off when in close proximity to the ground as in landing or to another aircraft as in aerial refueling. For such flight phases, the Level 1 requirement would have to be met with the SAS off. The statement in that spec paragraph which deals with "yaw damper ON or OFF" is merely an example of a selected configuration.

THE APPLICATION OF EQUIVALENT SYSTEMS

TO MIL-F-8785B

J. Hodgkinson - MCAIR

ABSTRACT

Equivalent systems allow analysis of augmented dynamics by building upon experience with unaugmented dynamics. Therefore, MIL-F-8785 requirements for augmented dynamics should likewise build upon existing requirements which describe essentially unagumented systems. This is simply achieved by retaining existing modal requirements, and adding a requirement restricting the equivalent delay caused by augmentation lags in both longitudinal and lateral dynamics.

Introduction

New data on the flying qualities of augmented aircraft have been analyzed using equivalent systems by a number of researchers. This is based on the generally held supposition that parameters such as equivalent frequency and damping adequately define the pilot's perception of the general response shape. This supposition has been upheld by preliminary data from a recent in-flight simulation, Reference 2. Available data also indicate that flying qualities problems due to frequency and damping deficiencies are indeed adequately predicted by these equivalent parameters and the requirements of MIL-F-8785B, Reference 1. This is covered more thoroughly in a forthcoming AGARD paper (Reference 3).

It is important to note that the great majority of our experience in analyzing augmented dynamics is with the equivalent system technique. A large number of studies have turned to order reduction as the first step in analysis of high order responses. References 4 through 15 are representative. These references discuss not only longitudinal short period flying qualities but also phugoid dynamics, approach power compensator effects, direct side force modes, direct lift blended dynamics, and lateral-directional fighter responses. Applications to V/STOL hover and transition effects are presently being carried out also. By using equivalents, flying qualities problems are parameterized for analysis and the insight gained with unaugmented aircraft is readily brought to bear on augmented

dynamics. Ideally therefore, augmentation effects as identified by equivalents should be treated by adding to the existing requirements on unaugmented dynamics. This paper discusses and arrives at a recommendation on what this addition should be.

Longitudinal Control System Lag Effects

Augmentation systems deliberately modify response transfer function denominator effects to obtain improved frequency and damping, for example. They also add actuators and sensors, prefilters, compensatory networks and structural mode filters. Digital systems add computation delays together with other components which produce additional lag (Reference 16). In-flight simulations (Reference 4, 5, and 17) have all shown that this lag can dominate the pilot's opinion. Equivalent systems to date have modeled this lag in two ways. First equivalent $1/T_{\theta_2}$ in the pitch dynamics can be allowed to seek artifically high values in the matching process and interpreted as higher equivalent n_{z_α}. This effectively treats mid-frequency lags. Second, a delay term is generally added. This approximates high frequency lags.

MIL-F-8785B(ASG) in 3.5.3 limits control system phase lag to a maximum value at the aircraft natural frequency. This effectively allows larger delays if the aircraft response is already slow, or assumes that a given delay becomes more objectionable as the aircraft frequency increases. This was based on DiFranco's in-flight data (Reference 4). This latter concept is tested in Figure 1, using the somewhat later data of Neal and Smith (Reference 5). Only configurations possessing good (i.e. Level 1) damping and frequency are shown, thereby attempting to remove their effects from the experiments. The expected trend of worsened rating with jointly increased delay and frequency does not appear.

Figure 2 is another attempted correlation using Neal and Smith's data, in this instance between delay alone and pilot rating. Little correlation is apparent. However, the data do not preclude the expected trend of worsened rating with increased delay, within the scatter apparently caused by changes in the other experimental variables. This suggests a technique which accounts for simultaneous variation of a number of experimental parameters. The technique chosen (and described in Reference 7) was stepwise multiple linear regression. This was used to generate pilot rating prediction formulae such as that of Figure 3. Note that now a reasonably accurate predictor of rating is obtained, and that the delay term produces a linear, additive rating degradation of one Cooper-Harper point for roughly each .05 seconds of delay (18.5 rating points per second).

Recently, in-flight data on augmented longitudinal dynamics in the landing approach (Smith, Reference 17) have become available. Figure 4 uses the equation developed from Neal and Smith's data to predict Smith's flare maneuver pilot ratings. Neal and Smith generally used a parallel feel system and Smith used a series system. However in spite of this the pilot ratings for the flare maneuver are broadly consistent with the up-and-away results. The equivalent delay due to the series feel system adds approximately one rating point to each estimate. This is indicated in Figure 4 by shifting the correlation line rather than each data point. (Ringland and Johnston noted this ability to predict a landing approach rating with the up-and-away equation in Reference 13. Since this was for a single configuration only, they deemed it to be fortuitous.) However, in contrast to Neal and Smith, Smith deliberately isolated lag effects. The results are shown in Figure 5. There is an apparent threshold in pilot's sensitivity to delay of about .07 seconds for good configurations, with a subsequent slope which is now around 35 rating points per second, double the value of the up-and-away prediction equation. This slope is also greater than the value of 25 obtained in a separate

regression of the landing approach data. For already unsatisfactory configurations, the eventual degradation is similar but the threshold is larger. The data for these unsatisfactory configurations tend to contradict the current requirement on phase lag since rating degradation with delay is greater for the slowest configuration than for the fastest.

Therefore the two most recent in-flight experiments on high order systems are not consistent with the earlier data which formed the basis for the MIL-F-8785 lag requirement. A possible explanation for this is that high stick force levels adopted in the earlier study precluded the aggressive maneuvering and demanding task which are necessary to discriminate configurations with control lags. It is further possible that these stick forces were chosen to ameliorate the response abruptness following an initial delay.

Lateral-Directional Equivalent Systems

Since the great majority of basic research on augmented aircraft flying qualities has been on longitudinal dynamics, lateral-directional problems must at present be tackled partly by extrapolation. Reference 18 summarizes experience with obtaining lateral-directional equivalents reliably from given high order dynamics. Briefly, when lateral-directional coupling is present, the dutch roll is best identified from the sideslip to directional control response and the roll mode from the roll rate (or bank angle) to lateral control response. This is somewhat different from the methodology which has evolved for the longitudinal dynamics, in which coupling is ignored, if this is necessary to obtain an accurate parameterization of the primary pitch response.

Preliminary data are becoming available from a joint USN/USAF/MCAIR equivalent systems simulation using the USAF/CALSPAN variable stability NT-33 (Reference 2). The flying qualities effects of lag in lateral dynamics for the landing approach were examined. The data suggest that the MIL-F-8785B(ASG) requirements on roll mode are adequate, but a new requirement is needed for lags. (Note that no data

were available as background to the current 3.5.3 requirements on lateral and directional lags.) Indications are again that a good configuration has a smaller threshold of rating degradation due to delay than a worse one. These thresholds are larger than for longitudinal dynamics. The subsequent sensitivity however is then roughly twice the value for longitudinal landing approach dynamics. These data have just become available as this is being written however, and in any case will be no substitute for the in-depth research which is needed on lateral-directional control system effects.

Recommendations for MIL-F-8785B Requirements

Longitudinal requirements - These should be stated in terms of equivalent systems. The matching process should retain $1/T_{\theta_2}$ at the basic value if possible; if not, the current $\omega_{n_{sp}}$ vs n/α requirement should be considered as a pitch criterion by calculating an equivalant n_{z_α} using the equivalent $1/T_{\theta_2}$ and the true speed. This is described and substantiated in References 3 and 7.

The particular effects of equivalent delay, the main topic of this paper, are adequately covered by substituting a Level 1 maximum of .1 seconds and a Level 2 maximum of .2 seconds for the current phase lag requirements of 3.5.3.

Lateral requirements - These also can be stated in terms of equivalent systems, as described and substantiated in Reference 18.

Delay effects can tentatively be covered by substituting a Level 1 maximum of .2 seconds and a Level 2 maximum of .25 seconds for the current requirements of 3.5.3.

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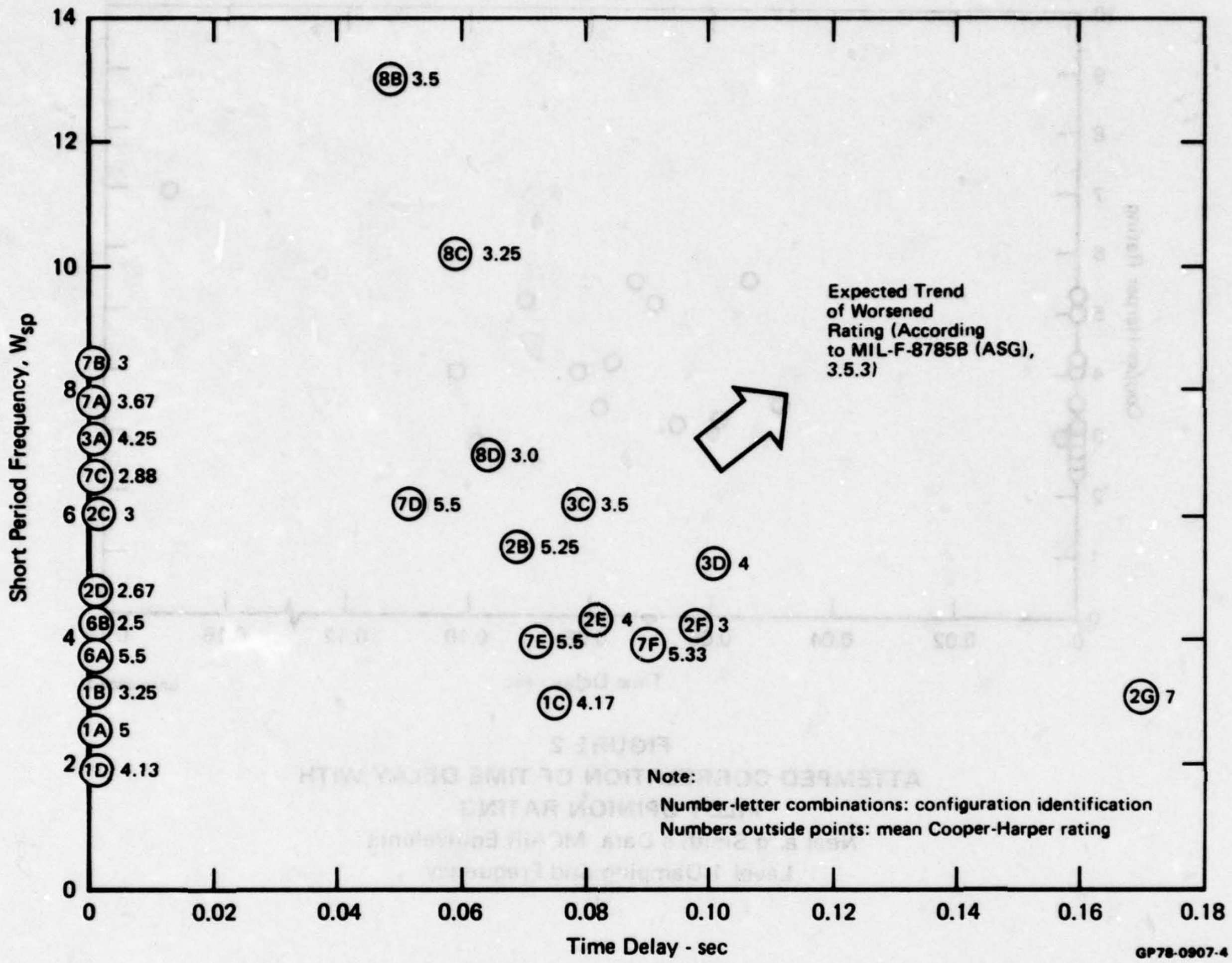


FIGURE 1

ATTEMPTED CORRELATION BETWEEN SHORT PERIOD FREQUENCY, TIME DELAY AND PILOT RATING

Neal and Smith's Data, MCAIR Equivalents, Level 1 Damping and Frequency

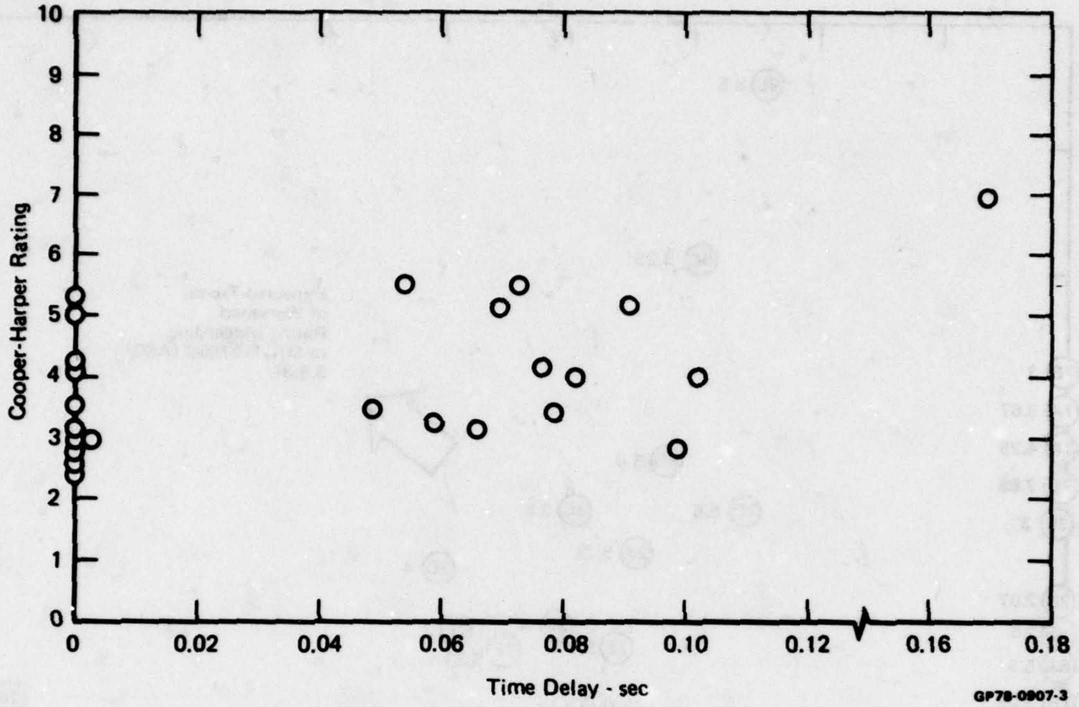
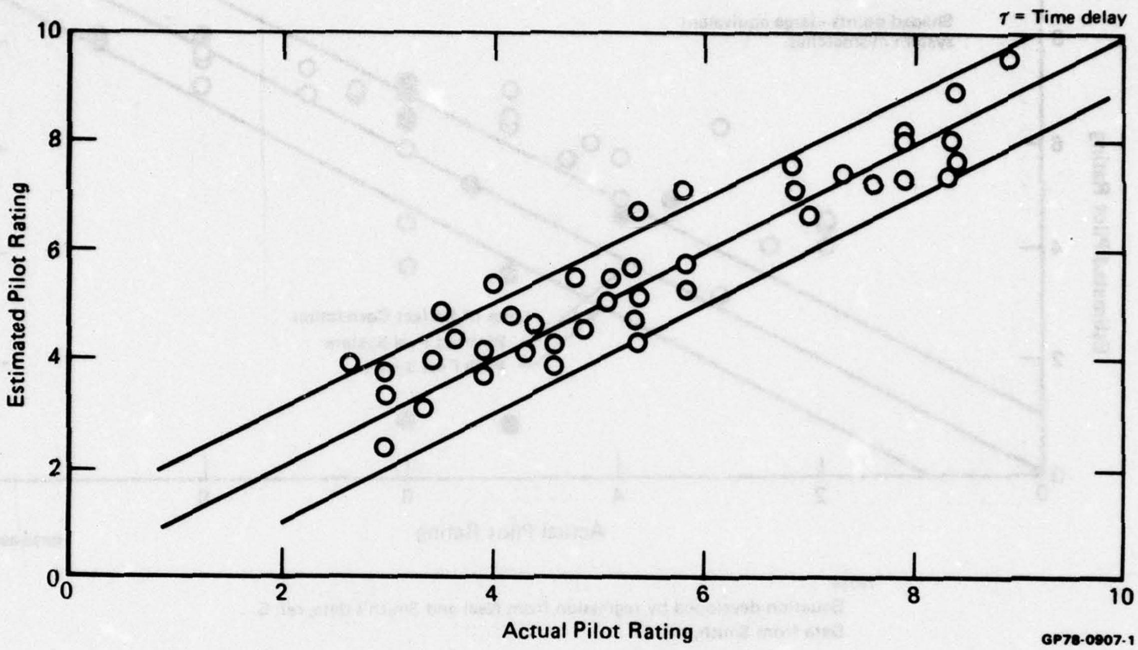


FIGURE 2
ATTEMPTED CORRELATION OF TIME DELAY WITH PILOT OPINION RATING
Neal and Smith's Data, MCAIR Equivalents, Level 1 Damping and Frequency

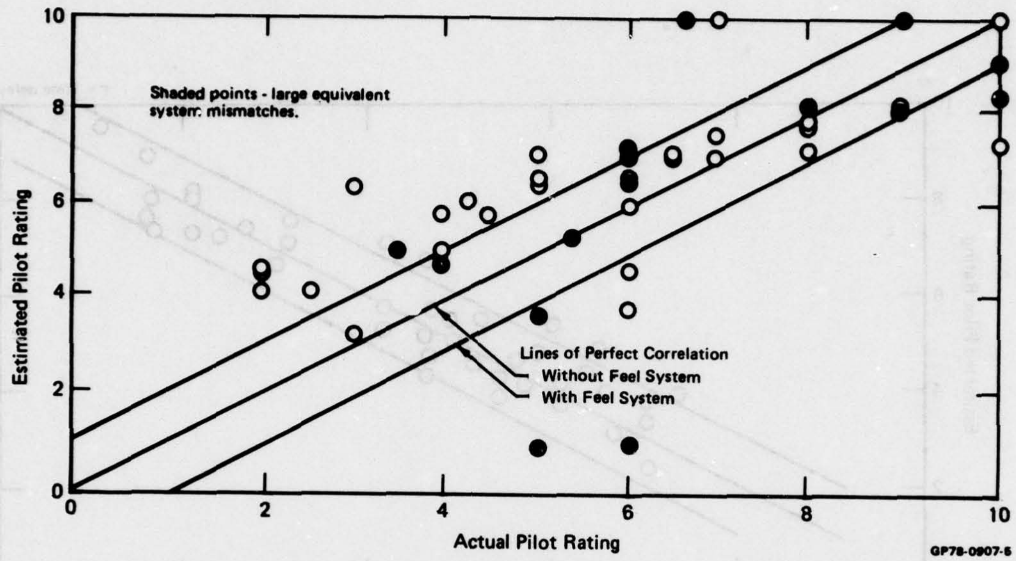


Note:

Equation developed by regression from Neal and Smith's data, ref. 5

Pilot rating = $5.16 + 18.50\,\tau + 0.56\,(L_{\alpha_e}/2\,\zeta_e\omega_e) - 0.61\,(\zeta_e\omega_e) + 0.02\,(\omega_e^2)$
(Cooper-Harper)

FIGURE 3
EXAMPLE OF PILOT RATING PREDICTION USING EQUIVALENT SYSTEM PARAMETERS



Note:

Equation developed by regression from Neal and Smith's data, ref. 5
Data from Smith, ref. 17

Pilot rating = $5.16 + 18.5\,\tau + 0.56\,(L_{\alpha}/2\zeta_e\omega_e) - 0.61\,(\zeta_e\omega_e) + 0.02\,(\omega_e^2)$
(Cooper-Harper)

FIGURE 4
PREDICTION OF PILOT RATING FOR LANDING APPROACH USING EQUATION DEVELOPED FROM UP-AND-AWAY SIMULATION

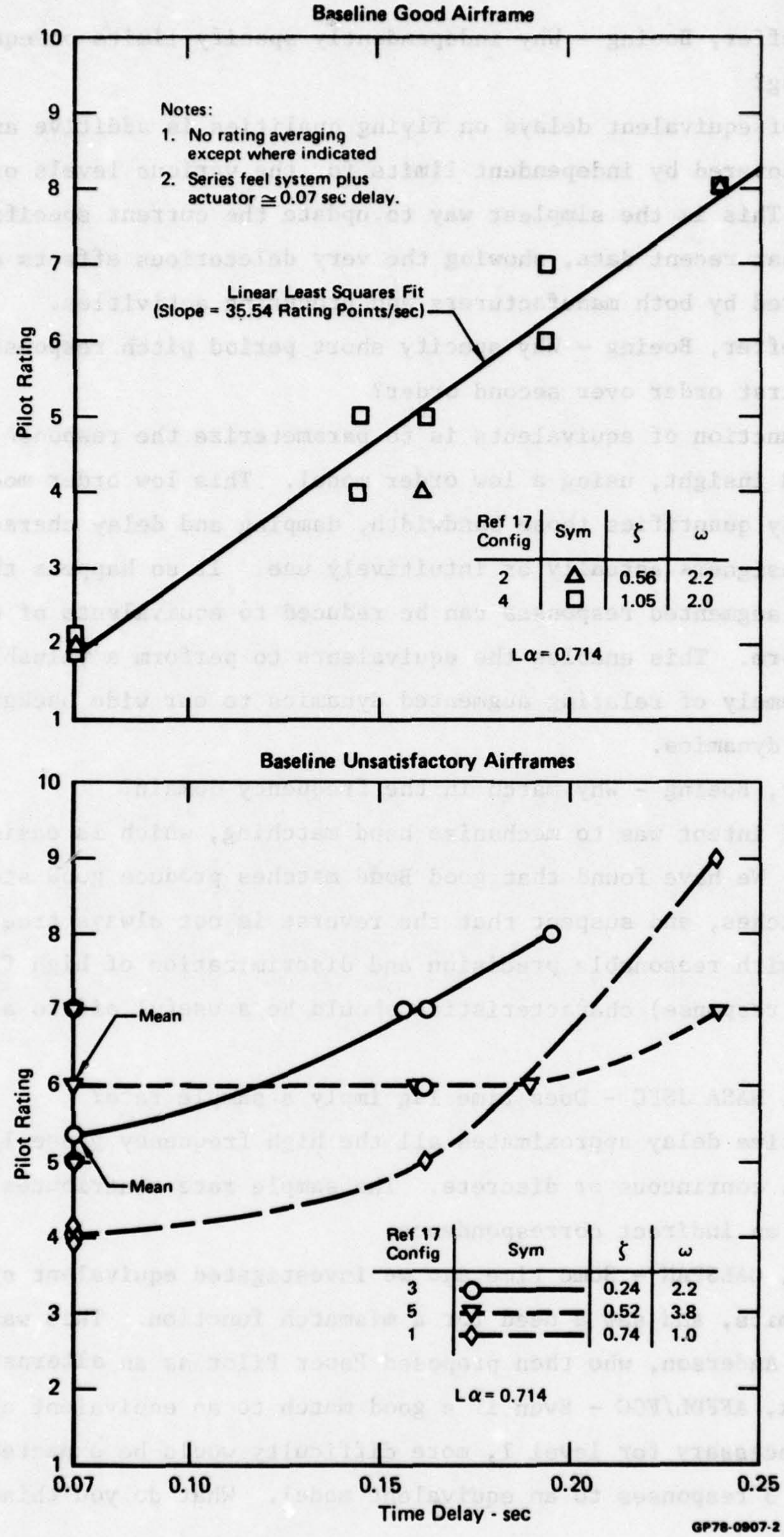


FIGURE 5

DEGRADATION OF PILOT OPINION RATING DUE TO EQUIVALENT TIME DELAY

LAHOS Data, MCAIR Equivalent Systems

- Dwight Schaeffer, Boeing - Why independently specify limits on equivalent transport lag?

  The effect of equivalent delays on flying qualities is additive and adequately covered by independent limits for the various levels of flying qualities. This is the simplest way to update the current specification to ensure that recent data, showing the very deleterious effects of lag, are considered by both manufacturers and procuring activities.

- Dwight Schaeffer, Boeing - Why specify short period pitch response as a classical first order over second order?

  The first function of equivalents is to parameterize the response for analysis and insight, using a low order model. This low order model approximately quantifies those bandwidth, damping and delay characteristics which all designers actually or intuitively use. It so happens that the majority of augmented responses can be reduced to equivalents of quasi-classical form. This enables the equivalents to perform a valuable second function, namely of relating augmented dynamics to our wide background with unaugmented dynamics.

- John Schuler, Boeing - Why match in the frequency domain?

  The original intent was to mechanize hand matching, which is easiest with Bode plots. We have found that good Bode matches produce good step time response matches, and suspect that the reverse is not always true. However, any method with reasonable precision and discrimination of high frequency (or initial response) characteristics should be a useful aid to a competent engineer.

- John Stigal, NASA JSFC - Does time lag imply a sample rate?

  Equivalent time delay approximates all the high frequency phase lags from all sources, continuous or discrete. The sample rate contributes lag, so there is an indirect correspondence.

- Chick Chalk, CALSPAN - Some time ago we investigated equivalent systems for V/STOL dynamics, and saw a need for a mismatch function. This was unacceptable to Ron Anderson, who then proposed Paper Pilot as an alternative.

- Bob Woodcock, AFFDL/FGC - Even if a good match to an equivalent classical system is necessary for level 1, more difficulty would be expected in fitting level 2 and 3 responses to an equivalent model. What do you think the prospect is?

  As you point out, "unmatchable" configurations have flying qualities problems.

However, the equivalent parameters for the level 2 and 3 cases we have analyzed are consistent with MIL-F-8785, while accepting the high mismatch. Analysis of data from the recent simulation on equivalent systems should answer some of our questions on mismatch.

- Chick Chalk, CALSPAN - What do you think of the boundaries for equivalent systems proposed by Mayhew for MIL-F-8785?

[See AFFDL-FGC - Working Paper, Proposals for Revising MIL-F-8785B, Vol. I, February 1978, pages 86 and 87].

They are too complex and unwieldy. Equivalent systems instead should be used to demonstrate compliance with all the current modal requirements, and an updated requirement on lags or delays should be added. This is consistent with available data and is an expeditious way of screening out:

(1) misleading 'dominant root' models

(2) configurations with large delays, which are unsafe.

SECTION V

SESSION 3: THE PROPOSED REVISION TO MIL-F-8785B, AND COMMENTS

A SUMMARY OF VALIDATION REPORTS FOR MIL-F-8785B

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Introduction:

In the early 1970's, through a series of contracted efforts we examined the validity and completeness of the new Military Specification for Flying Qualities of Piloted Airplanes MIL-F-8785B (ASG) by application to existing airplanes. (Note: the F-4 was compared with 8785A, which was essentially the same.) The efforts consisted of paragraph-by-paragraph comparisons of 8785B requirements with the documented characteristics of aircraft currently flown by the U.S. military and its allies and with the available pilot comments and ratings. Aircraft selected for the study were chosen based upon the following criteria:

i) a well documented history with a sufficient amount of data to perform quantitative analysis
ii) a sufficient number of operational flight hours
iii) aircraft representative of a specific class

Based upon these criteria the following aircraft were chosen:

Ref.	Airplane	Class	AFFDL TR
1.	F-4(B,C,D,E,J) Phantom	IV	70-155
2.	F-5(A,B)/T-38A Talon	IV	71-134
3.	P-3B Orion	III	72-141
4.	C-5A Galaxy	III	75-3

The author will summarize but will not comment on the validity of any recommendations proposed in the reports. In order to assess validity it is suggested that the reader consult the background discussions in the reports cited.

Evaluation of Requirements

This section presents the comparisons of the four airplanes to the requirements of the Flying Qualities Specification (Refs. 5 & 6). Specification paragraphs of Sections 1 and 3 for which recommendations have been suggested will be annotated in the following manner:

** - addressed by the proposed 1978 revision (Ref 7) to 8785B

* - "Significant" items that warrant consideration but have not been addressed by the present proposed revision

Significant recommendations are considered to be those:

1) resulting from marked inconsistency of pilot ratings and comments and the requirements.
2) for which two or more airplanes have similar discrepancies in the requirements.

If conflicting recommendations have been proposed for two airplanes in the same class, the author chose not to present the recommendations. All recommendations have been substantiated through pilot ratings/ comments, data, or both. Background discussion or aircraft identification will be given where it is considered necessary or helpful for clarification. Notes will be made where a proposed revision conflicts with or does not incorporate the recommendations proposed by the reports.

Requirements and Recommendations

1. - SCOPE AND CLASSIFICATION

1.1 - Scope*: It is necessary to recognize and deal with the need for coordination between MIL-F-8785B and MIL-F-83300, the V/STOL flying qualities specification (Ref. 3).

1.2 - Application**: The wording of the requirement implies that deviations from the specification are allowable. It should suggest that special requirements, whether additional or alternate, may be specified by the procuring activity (Ref. 2).

1.4 - Flight Phase Category: Missions of specific airplanes differ enough to make any listing of discrete flight phases possibly ambiguous, incomplete, or superfluous. Rewrite 1.4 and expand the background document in support of 1.4 to include the Flight phases which are considered in each category (Ref. 3).

1.5 - Levels of flying qualities**: The verbal discussion for the various levels of flying qualities should be placed in the background document (Ref. 3). (Note: a major rewrite has been proposed for Section 1 in 1978 revision.)

Section 3 - Requirements

3.1 - General requirements

3.1.1 - Operational missions**: A good, clear and concise mission definition is lacking (Ref. 1,3,4). Special maneuvering requirements should be addressed and the spectrum of usage should further be defined (Ref. 1).

3.1.2 - Loadings: The requirement to define loadings that will exist for each flight phase is not realistic for certain aircraft with a wide spectrum of operational missions and with possible variations of profiles within a mission (i.e. Ref. 3 on ASW). The Contractor should define the envelopes of c.g. and corresponding weights that will exist throughout the operational mission (Ref. 3). Also, clarification is needed on the data to be used to define the critical loading for corresponding normal and failure states.

3.1.6.1 - Airplane Normal States: The procedure for selecting critical values of such items as gross weight, c.g. and moment of inertia is obscure. Reference should be made to requirements for loading data (3.1.2), moments of inertia (3.1.3), external stores data (3.1.4) and the criteria of 4.2, the airplane states (Ref 1). Also, it should not be necessary to tabulate all the multitude of possible store loadings (Ref. 2). There is a difficulty in defining all states (i.e. loadings, c.g., etc.)(Ref. 3); consequently it is felt the required information should be defined and tabulated with the guidance and approval of the procuring activity (Ref. 2 & 3).

3.1.6.2 - Airplane Failure States: Revision is needed to reduce the task of compliance. The contractor should define and tabulate those those failure states which have a significant effect on flying qualities (Ref. 1). There is also the need to define realistic requirements for probability of encountering degraded flying qualities and tailoring these by considering the machine and its mission (Ref. 3).

3.1.7 - Operational Flight Envelope: The minimal wording of the specification doesn't ensure the understanding that V-n diagrams are required (Ref. 2). There is potential conflict in the application of

the specification Table I for flight phase CR, RT and D: this occurs when a power-off low-speed stall-related boundary intersects an altitude boundary established by the rate of climb capabilities at a specific thrust. The power or thrust requirement differs and presents a quandry. Guidance whould be given when a conflict occurs (Ref. 3).

3.1.8.4 - Service load factors: Identify the specific configurations for which service load factors as a function of speed must be constructed (Ref. 2). Also address the airplanes capabilities and not the minimum angle requirement. Onset of stall warning is the cue by which the pilot determines the proximity to stall and should be the boundary of the service flight envelope (Ref. 3).

3.1.9.2.1 - Minimum permissible speed other than stall speed**: Airspeeds below stall speed should be allowed for special conditions, especially for class IV aircraft with air-to-air combat requirement to permit exploitation of its agility against an adversary. The following stipulation is placed on this recommendation, that pilot safety is not jeopardized and dangerous flight conditions will not arise (Ref. 2).

3.1.10.2.1 - Requirements for specific failures: The definitions of the specific failures are not sufficiently clear to make the requirement comprehensible. There is a need to perform failure state analyses in the design phase and to determine which support systems, subsystems and components affect the flying qualities. Specific failures that are known to affect the flying qualities should be defined by the contractor with the approval of the procuring activity. Also, the inclusion of reliability requirements in the present content of specification encumbers its application and tends to produce an oversimplified approach to reliability analysis (Ref. 3).

3.1.10.3.2 - When Levels are not specified: When Levels are not specified, retention of this requirement in the content of the current specification necessitates further development of some paragraphs in 3.1.10.3.2 for Level 1, Level 2 and Level 3 requirements. The broad nature of 3.1.10.3.2 imposes unduly severe requirements in some cases such as 3.6.1.4 Trim system irreversibility (Ref. 3).

3.2 - Longitudinal Flying Qualities

3.2.1.1 - Longitudinal static stability**: The addition of a statement, "the combined effects of centering, breakout force, stick force, stability and force gradient shall not produce objectionable flight characteristics" is desirable. Also allow neutral gradients for Level 1 category A and B. Define a maximum positive force gradient limit (Ref. 1). There is an incompatibility in the requirement for a fixed trimmer/throttle position for 1g and constant altitude (Ref 2,3, 4). There is no way to conduct tests at constant altitude unless power is varied.

3.2.1.1.2 - Elevator control force variations during rapid speed changes: Reword to exclude manual trim systems at Level 1 Cat. A for Class III aircraft which are required to accelerate and decelerate rapidly over large speed ranges for tactical maneuvers (Ref. 3).

3.2.1.2 - Phugoid stability*: The requirements are felt to be too stringent. Lower Level 2 boundary for ζ_{PH} from 0.1 to $\zeta_{PH} = 0$ (Ref. 1). It was noted that Level 3 values were receiving Level 1 and Level 2 pilot ratings. Three of four aircraft experienced the same recurring problem. They were getting different acceptable values for ζ_{PH} from those considered acceptable in the requirement (Ref. 1,3,34). Subject to the approval of the procuring activity, relaxation of the Level 1 requirement should be permitted when period is greater than 30 seconds (Ref. 4).

3.2.2.1 - Short-period response: Clarify the meaning of coincident Level 2 - Level 3 boundaries in paragraph 6.7.2 of specification. Make sure reference to 3.2.2.1.1 in background document is placed in 3.2.2.1 which discusses the problem (Ref. 3).

3.2.2.1.2 - Short-period damping*: Pilot comments indicate that interactions of various parameters have an effect on pilot opinion, and the extent of the influence of ζ_{sp} alone on the quoted rating is sometimes unclear. Further investigation is warranted. NATC reports suggest differences up to .05 in ζ_{sp} are below pilot threshold, yet .05 is the total bandwidth of the Level 3 band in Category B. With

this in mind Level 2 and 3 bandwidths seem impractically narrow (Ref. 1). The specified minimum ζ_{sp} of .25 for Level 2 Catagory A & C flight phases for Class IV is not considered to be the lowest acceptable minimum value (Ref. 2). Additional data needs to be obtained for Class III airplanes to support or revise the Category A Flight Phase requirement. For Category B the lower bounds of Level 1 should be relaxed for Class III airplanes contingent upon procuring activity approval. For Class III reduce the lower bounds of figure 2 for Level 1 by 10% (Ref. 4).

3.2.2.2.1 - <u>Control forces in maneuvering flight</u>**: Relax Level 3 minimum boundary to 2.0 lb/g (Ref. 1). Average gradient needs to be defined (Ref. 2). For wheel control airplanes required to maneuver tactically or designed to be flown one-handed, the upper and lower force limits should be relaxed (Ref. 3).

3.2.2.2.2 - <u>Control motions in maneuvering flight</u>: Pilots commented that a defined upper limit for Class III Level 1 flying qualities to control force per inch might result in lower F_s/N values being found acceptable during tactical maneuvers (Ref. 3).

3.2.2.4 - <u>Longitudinal Pilot Induced Oscillations</u> **: A definitive criterion which would catch an PIO problem in the design stage is certainly desirable. The qualitative form of the requirement as it presently stands is considered insufficient as a specification to ensure no PIO tendency (Ref. 1 & 2).

3.2.2.3.1 - <u>Transient control forces</u>: Relaxation is needed for Class IV airplanes in CO Flight Phase at the discretion of the procuring activity. This relaxation should not be allowed to cause any adverse effects such as a tendency to PIO (Ref. 1). Also 6 lb/g is felt to be too low for wheel control Class III airplanes. Class III should be excluded from coverage under 6 lb/g requirement. A more complete survey of the sudden pullup characteristics of Class III airplanes should be conducted; this should be the impetus for establishing a new minimum F_s/n for Class III airplanes (Ref. 3).

3.2.3.4 - <u>Longitudinal Control in Landing</u>: Requirement should indicate whether the airplane should be in or out of ground effect

when compared with the requirement (Ref. 1). Change 3.2.3.4/3.2.3.4.1 to provide similar coverage during landing as 3.2.3.3 provides during take off (Ref. 3).

3.2.3.5 - Longitudinal control in dives**: Requirement appears to excessively overlap 3.6.1.2. For Class III consider wheel control aircraft with manual trim systems as a center-stick-controller aircraft for the longitudinal force characteristics (Ref. 3 & 4).

3.2.3.7 - Longitudinal control in sideslips: This requirement should be rewritten to provide adequate coverage for multi-engined aircraft with asymmetric drag or thrust or the need to crab and de-crab in cross-winds during take off, approach, and landing (Ref. 3).

3.3 - Lateral-directional flying qualities

3.3.1.1 - Dutch roll**: There is a general disagreement with pilot rating and specification requirements for Class III airplanes. Additional data should be obtained to substantiate any revision (Ref. 3 & 4).

3.3.1.2 - Roll mode*: There is a need for a study to specifically review the roll mode time constant and the effect of roll-rate damping augmentation which has been utilized in aircraft design (Ref. 2). Ref. 3 proposed two new roll rate damping requirements based on 1) the quality of response to a pilot input and 2) the adequate suppression of lateral gust disturbances. For airplanes in which personnel are located at a considerable distance from the principal roll axis the requirement of 3.3.1.2 should be reasonably relaxed. Additional information is needed to support this Class III requirement (Ref. 4).

3.3.1.3 - Spiral stability**: The entire question of acceptable spiral mode characteristics for tactical instrument flight should be reviewed (Ref. 3).

3.3.2.4 - Sideslip excursion*: A clear definition of $\Delta\beta$ is desirable and guidance as to which time to use when calculating "K" is needed. Directions on application of the paragraph are not stated (Ref. 3 & 4). Uniform applicability of the requirement to all classes of aircraft is questionable. The requirement for a bank-angle change of 90° is excessive for Class III. Aileron command should be held in long enough

to establish (ϕ_t) command and β (Ref. 4).

3.3.3 - Lateral-direction Pilot Induced Oscillation*: Little or no work has been conducted in the industry to establish criteria to evaluate lateral PIO quantitatively. It is considered essential that a method be derived and quantitative evaluation be specified. It is recommended that research work be conducted in this field to establish a quantitative specification (Ref. 2).

3.3.4 - Roll control effectiveness**: For Class IV-L and C aircraft the Level 1 minimum time to bank to 30° requirement should be relaxed to 1.3 seconds and the lower Level 3 boundary for the time to bank to 30° should be relaxed to 2.8 seconds (Ref. 1). It appears the requirements of 3.3.4 for Class III airplanes have been arbitratily selected. C-5 results indicate large, heavy transport airplanes can have satisfactory performance (Level 1 from pilot rating and comments) without meeting Level 1 requirements. Class III roll control effectiveness requirements are considered to be too stringent, and further investigation and re-evaluation are required (Ref. 4).

3.3.4.1.1 - Air to air combat**: Level 3 requirement could be relaxed provided that the resulting roll performance is adequate to break off an engagement and escape from an opponent (Ref. 1). Also allow pilots to use rudder pedals in extreme α and β situations where he is confident in his ability to maintain control. A feasibility study of relaxing restriction on rudder-pedal-augmented rolls should be made (Ref. 2).

3.3.4.2 - Aileron control forces*: For Class IV specify a Level 2 band separating Level 1 and Level 3 requirement. According to figure 1 there is a markedly different standard of mission between Level 1 and Level 3 requirements. This should be recognized by the inclusion of a Level 2 band (Ref. 1). The upper limit force characteristics of Table X should be re-evaluated for Class III airplanes that required the pilot to fly "one hand on the wheel" in order to perform tactical maneuvers (Ref. 3).

3.3.4.3 - Linearity of roll response*: Research is needed to investigate, identify and establish the objections (through pilot rating and comments) to various degrees nonlinearity in response. It is

suggested that a quantitative requirement can be specified in terms of maximum variation in a local slop. from a mean gradient (Ref. 2).

3.3.7 - Lateral-directional control in crosswinds: For Class III, rewrite the Level 3 requirements since they appear to be inconsistent with Level 1 and Level 2 flying qualities. Add requirement for Class III to demonstrate Level 2 flying qualities in crosswinds up to 40 knots instead of 30 knots (Ref. 3). [Note, however, that an extensive study of crosswind requirements was made for MIL-F-8785B.]

3.3.7.3 - Taxiing wind speed limits*: Requirement could impose an engine design penalty for airplanes with fan engines, which is probably not intended. A review of the impact of the requirement should be undertaken on other currently operating airplanes which employ high-powered fan engines. The C-5 fan engines operating in crosswind and tailwind conditions up to 30 k ots experience no performance degradation. But above 30 knots reduced power settings must be observed. At 45 knots there is insufficient power generated in order to taxi. The recommendation is to establish taxi wind requirement as amargin above the required crosswind components on the basis of operating experience (Ref. 4).

3.4 - Miscellaneous flying qualities

3.4.2.4 - Stall recovery and prevention: For Class IV airplanes, since three aerodynamic controls are allowed to be used in stall recovery there is no apparent reason why throttle manipulation should not be allowed (Ref. 2). For Class III airplanes further studies should be accomplished to establish acceptable limits for excursions in pitch, roll and yaw (Ref.3).

3.4.3 - Spin recovery: Consideration should be given to revising spin-demonstration requirements to include spin susceptibility (Ref. 1). (This was done in the amendments to MIL-F-8785B).

3.4.4 - Roll-pitch-yaw coupling: Class IV airplane requirements place emphasis more on structural limits then on resulting dangerous flight conditions due to rolling at high angles of attack (especially in the $C_{L_{MAX}}$ region). When rolled at $C_{L_{MAX}}$, some aircraft will spin

or uncontrollable motion will occur. Also replace "$.8n_L$" by "$C_{L_{MAX}}$" or $.8n_L$, whichever comes first (Ref. 2). For Class III aircraft it is considered not a definitive requirement. Inlcusion of the 100% structural demonstration compliance with requirement is somewhat incompatible. This should be performed at 80% of the structural capability of the airplane. Also, heavy transports may not be allowed to roll 120° (90° is felt to be more applicable). Abrupt, uncoordinated rolls at load factors from 0g to $.8N_L$ may not be possible for large CIII transports.

3.5 - Characteristics of the primary flight control system

3.5.2.1 - Control centering and breakout forces*: Relax breakout force limits for Class IV as follows; elevator for Level 1 should be 4 lb; set aileron upper limit for Level 1 to 2 lb and 5 lb for Level 2; and increase rudder to 14 lb (Ref. 1 & 2). For airplanes with cable systems lower breakout forces and force gradients should be considered (Ref. 4). A revision to descirbe a specific technique to be used for measuring breakout point is desired (Ref. 3 & 4). It has been suggested that the breakout forces should be the force measured at the first significant movement of the control surface (Ref. 3). Also it is desirable to measure forces inflight (Ref. 4).

3.5.2.2 - Cockpit control free play: There is a need for a quantitative requirement based upon real-world operational experiences. Also it is desirable to establish maximum allowable normal and failure-state free-play limits for wheel and stick control systems for all Classes of airplane (Ref. 3).

3,5,2,3 - Rate of control displacement: Reword to specifically include emergency flight operations where the combinations of a large control displacement accompanied by high controller force limits the ability of the pilot to maneuver (Ref. 3).

3.6 - Characteristics of secondary flight control system**:

3.6.1 - Trim systems**: Relax the requirement for asymmetric loadings provided the operational effectiveness of the aircraft is not unduly compromised (Ref. 1). Allowable aileron force for Level 3 after failure should be increased above allowable breakout force (Ref. 3). [Note: recommendation differs from revision in Ref. 7.]

3.6.3.- <u>Transient and trim change</u>**: The trim shange is not dealt with. Requirement should cover objectionable transient nature of trim changes. Also add a statement ot prohibit excessive control forces or other objectionable demands on the pilot (Ref. 1). (Note recommendation differes from revision in Ref. 7.)

3.7 - <u>Atmospheric disturbances</u>

3.7.5 - <u>Application of the turbulence models in analysis</u>*:

Requirement does not specify criterion for flying in turbulence (Ref. 1,2,3,4). (This is discussed in Volume II of Reference 7.)

<u>Conclusions</u>

In general, the authors of Refs 1-4 consider MIL-F-8785B to represent a substantial improvement over past specifications with regard to requirements definition, format, and overall clarity. Reference 1 considers the following areas of the specification as candidates for further investigation:

1) Longitudinal Short Period damping ratio (3.2.2.1.2)
2) Longitudinal Pilot induced oscillations (3.2.2.3)
3) Roll mode time constant (3..31.2)
4) Spin recovery (3.4.3)
5) Control system mechanical characteristics (3.5.2)
6) Engine control and response characteristics (3.6.2)
7) Quantitative requirements on atmospheric disturbances (3.7)

in addition to the above, the following general topics are considered to be in need of further study:

8) Practicability of the General Requirements section
9) Specification of parameters relavant to aircraft with stability augmentation systems
10) Effect of interaction of 'good' and 'bad' parameters on overall mission capability.

Finally the authors of Reference 1 reached the following general conclusions:

1) The flying qualitiesspecification is a considerable improvement over its predecessors
2) The intent of the General Requirements section is understood;

however, it presents an obscure and idealistic definition of a mammoth task.

3) In a number of cases overly conservative quantitative requirements have been specified when substantiating data are absent, scant or inconclusive.

4) A number of requirements have limited applicability to aircraft with artificial stability augmentation systems.

5) The importance of using a pilot opinion rating method such as the Cooper-Harper scale in testing for compliance with qualitative requirements, cannot be too strongly emphasized; the authors in many cases found assigning even a level of flying qualities to a qualitative remark was difficult or impossible.

6) The assessment of 'poor' flying qualities is difficult; in this connection Level 3 is often ill-defined.

Reference 2 yielded the following general conclusions:

1) the specification represents an outstanding improvement over past specifications

2) the two most pertinent new requirements needed to be expanded for more comprehension are:

    a) the "Airplane Failure States" (by including guidelines and sample approaches to provide evaluation methods for contractor guidance when comparing or designing airplanes to this specification).

    b) the "Atmospheric Disturbances" (requirements should be defined and should include quantitative values for compliance levels).

Reference 3 gave no formal overall conclusion, only the conclusions present in the comparison of each paragraph. It is this author's personal opinion after reviewing the report that Reference 3 agrees with the general conclusion that the specification is an improvement over the past specifications. Reference 4 presented the following conclusions:

1) generally, the data compared favorably with the specification except in certain sections where the requirements appear to

have been based primarily on medium and light weight airplane data.

2) based on the data, the following sections are far too stringent for Class III airplanes.

   a) Roll mode (τ_R) (3.3.12)

   b) Sideslip excursions (3.3.2.4)

   c) Roll control effectiveness (3.3.4)

3) additional data from Class III heavy aircraft be gathered to substantiate or revise the requirements in the following sections:

   a) Phugoid stability (3.2.1.2)

   b) Short period response (3.2.2.1)

   c) Control forces in maneuvering flight (3.2.2.2.1)

   d) Lateral-directional oscillations (dutch roll) (3.3.1.1)

   e) Roll mode (τ_R) (3.3.1.2)

   f) Sideslip excursions (3.3.2.4)

   g) Roll control effectiveness (3.3.4)

   h) Resistance to loss of control (3.4.2.2.1)

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Col. Shall, 4950th Test Eing: What is meant by your comment to define "$\Delta\beta$ and the time to K" in 3.3.2.4?

Answer: $\Delta\beta_{MAX}$ is defined in MIL-F-8785B, but not $\Delta\beta$. Also, there is a need to define an appropriate time for calculating "K".

Don West, Boeing: Does the Air Force really have a need to land in 40 knots cross wind?

Answer (from Moorhouse, AFFDL): What is presented in the paper is the recommendation in one of the validation reports. A large study of crosswinds was done for MIL-F-8785B; we do not plan to revise those requirements.

Discussion and Status of the Proposed Revision (1978) to MIL-F-8785B

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ABSTRACT

This paper discusses the the proposed revisions to MIL-F-8785B contained in the February 1978 Working Paper distributed for industry review (Reference 1). Comments and questions received from various reviewers are used with second thoughts by the authors to indicate the status of the revision effort. The intent is to clarify the proposed changes and/or their rationale.

I. INTRODUCTION

The current revision is the result of work done in the Control Dynamics Branch from 1973 through 1975. Various working papers were collected for internal review in early 1976, contributed by the following people:

J. Callahan, Maj USAF (currently assigned Columbus AFB, Miss.)

J. Lockenour (currently with Northrop Corp.)

D. Mayhew (currently with Draper Labs.)

D. Moorhouse

R. Quaglieri (currently with AFFDL/FGD)

M. Sanders, Capt USAF (currently assigned England AFB, La.)

R. Woodcock

During the summer of 1977 the three authors of this paper reviewed the collection of working papers, revising and correlating the proposals into a single document. Following preliminary coordination meetings with the Navy (Dec 77) and ASD (Jan 78) the current working paper (Ref. 1) was issued for industry review. A primary object of the symposium in general and this paper in particular is to facilitate the review process. It is planned to draft an Amendment 3 to MIL-F-8785B, based on Reference 1 and the results of this symposium before the end of 1978.

II. DISCUSSION

II.1 General

A summary of the proposed revisions is given in Figure 1. It is acknowledged that other requirements also need revision, but those will not be discussed here. The second object of the symposium is to identify these remaining deficiencies and to discuss future requirements. Some of the revisions are general in nature and apply to various paragraphs. The terms "elevator," "aileron" and "rudder" have been changed to the more general notation of "pitch," "roll" and "yaw control". Also, requirements on control surface deflection have

been deleted in favor of requirements relating aircraft response to pilot control input. Both of these changes admit the possibilities of advanced flight control systems - additional control surfaces and non-classical control responses. Each of the specific proposed revisions will now be discussed in order.

II.2 Scope and Classifications

1.1 Scope: The proposed definition mentions more explicit factors: inflight and on the ground, manned airplanes, speed at or above V_{con} Also, in response to some industry suggestions, material which 8785B places in 6.1 has been incorporated here to give it more prominence.

1.2 Application: The changes are an attempt at clarification, not considered to involve any change in intent of the specification.

1.5 Levels of Flying Qualities: MIL-F-8785B contained a detailed turbulence model, the most common use of which has been in piloted simulation. However, there are few specific requirements on the use of the model, or on the effects on flying qualities of atmospheric disturbances in general. The revisions proposed here do not redress that deficiency. We have attempted, however, to recognize explicitly that atmospheric disturbances do affect flying qualities. It will be seen that a number of considerations bear on the form this change should take.

Chalk, et. al., discuss ways of correcting this deficiency (Reference 2), specifically in piloted simulation. They recommend that a pilot "fly" the aircraft in smooth air, light-to-moderate (i.e. most probable) turbulence and severe turbulence. The pilot would be informed of the expected frequency of encounter of the different turbulence values and would then give a composite or overall rating. This is felt to be still an indirect way of evaluating the effects of disturbances, requiring subjective pilot judgment; such judgments should be minimized (Reference 3).

In the proposed revision of this section, repeated here as Figure 2, the intent was to account for the effects of atmospheric disturbances in an explicit manner. If the pilot is instructed not

to compensate mentally for the effects of the disturbances in his rating, then we have the basic use of the Cooper-Harper rating scale (Reference 3): the pilot is rating a given aircraft configuration to do a particular task in a certain atmospheric environment. In that respect, then, we would expect increasing disturbance magnitudes to worsen task performance and/or increase pilot workload to accomplish the task. These are the same as the qualitative effects of degrading basic airplane characteristics, heuristically leading to the form of the proposed revision given in Figure 2. This figure also shows the relation of this definition to Cooper-Harper ratings. The amount of allowable degradation at a given Level has an essentially subjective basis; for certain aircraft the procuring activity may desire to change that aspect of the Level definitions. It seems better for such judgments to be made by the Project Office than by individual evaluation pilots.

Now, let us consider an airplane design which is clearly adequate for its intended mission, i.e. "Level 1", and let us also consider the landing Flight Phase in the Operational Flight Envelope. In smooth air the task of landing this hypothetical aircraft should yield a Cooper-Harper rating better than 3.5. The rating should also remain better than 3.5 in disturbances up to LIGHT.* With greater, MODERATE* disturbances, glideslope tracking might degrade or pilot workload increase and the pilot rating for Level 1 would be allowed to degrade commensurately, but no worse than 6.5. Similarly, in SEVERE* disturbances the rating for Level 1 could degrade to worse than 6.5 but not beyond 9. This last statement is equivalent to requiring that a "good" or Level 1 airplane can be landed in something like a thunderstorm with reasonable confidence that the pilot will not lose control. To the authors this progression is logical and consistent with basic

*The terms LIGHT, MODERATE and SEVERE are intended to correspond to specific probabilities; 10^{-1}, 10^{-3}, and 10^{-5} are suggested, as defined in detail in Section 3.7. Such a footnote should be added to the proposed revision of 1.5.

flying qualities definitions. For an Aircraft Normal State, in the Operational Flight Envelope, the usual connotation of Level 2 and 3 numerical pilot ratings as requiring improvements would not necessarily apply to such effects of disturbances. A similar progression of the effects of disturbances is shown for airplanes which have less than Level 1 flying qualities in smooth air. In this case we have combinations of probabilities where the requirement is that, e.g. although a landing may be aborted, control must be maintained to make a go-around, "fly out of the disturbance."

In acknowledging the degradation in flying qualities due to atmospheric disturbances, we do not need to require that the airplane characteristics remain unchanged. Changes can occur due to basic non-linearities, augmentation saturation, etc. The point is that the proposal limits this degradation - it does not require degradation. This is consistent with the whole philosophy of MIL-F-8785B in presenting minimum acceptable requirements. The revision should not dissuade the use of gust alleviation or ride-smoothing systems - as now, the advantages and disadvantages of such systems must be weighed for each particular application. The procuring activity always has the option of increasing these minimum requirements, such as requesting all-weather capability. This could be achieved by a modified requirement, as illustrated in Figure 3. This example puts the design objective as maintaining basic flying qualities in disturbances up to values corresponding to a probability of 10^{-3}.

A few last comments on the proposed changes to 1.5 - they are intended to be more a clarification than an amplification of requirements, they address recognized questions of interpretation. For piloted simulations, the problems of simulating turbulence, especially the higher intensities, are acknowledged. The acceptability of pilot ratings is a function of the intensity of turbulence, wind shear, etc, and the proposed revisions define this trend. Analytical evaluation of flying qualities is unchanged pending the development of quantitative criteria for aircraft response to disturbances. It is suggested

that the proposed revisions form a framework for the development of such quantitative criteria. These could take the form of correlating pilot rating either with aircraft responses at the different levels of disturbance or with calculated pilot compensation to keep the responses within acceptable levels. Lastly, it is believed that evaluation of flying qualities in flight test will be aided, not hampered, by the proposed revisions. The proposal could be used to assess correctly a flight that "finds" turbulence (using weather information for the location). There is no requirement to fly in thunderstorm turbulence to demonstrate compliance with MIL-F-8785B, nor will there be a requirement to demonstrate compliance with all of the proposed 1.5 in-flight test. Flight test requirements will be determined by the procuring activity, as they are now.

This extended discussion of the proposed 1.5 was prompted by comments that ranged from unequivocal disagreement, through questions, to agreement.

II.3 <u>General Requirements</u>

3.1.1 <u>Operational Missions</u>: The proposed changes are an attempt to clarify the intent of the paragraph.

3.1.3 <u>Moments and products of inertia</u>: A semantic change to include products of inertia (if the paragraph is necessary).

3.1.9 <u>Permissible flight envelopes</u>: This proposed change will require the contractor to define the permissible flight envelope, in order to avoid any undue restrictions. It is recognized that this change is primarily oriented to advanced fighter configurations, whereas transports are likely to use the same limiting factors as presently in MIL-F-8785B. The present requirements could, therefore, be included in a back-up document (an updated version of Reference 4).

3.1.10 <u>Application of Levels</u>: The changes proposed for this paragraph are to make it consistent with changes in 1.5 and 3.4.11. Acceptance of the changes will therefore depend on acceptance of the other paragraphs.

II.4 Longitudinal Requirements

3.2.1.1 Longitudinal static stability: This proposal explicitly allows zero control gradients with artificial speed stability and, more significantly, also static instability for Level 3. For Levels 1 and 2 the basic requirement remains that "there shall be no tendency for airspeed to diverge aperiodically...". As stated in Reference 1, zero control gradients were not intended to be excluded by MIL-F-8785B. The second item is just to catch up with the facts represented by the success of the F-16 and many simulation results ground-based and flight. The proposals have so far received qualified approval.

One reason cited for the conservatism of the present requirement is to allow margin for design error and for operational growth. The proposed relaxation thus places more stringent responsibilities on both designers and users.

3.2.2.1.3 Higher order dynamic systems: This new paragraph was proposed for certain configurations as an alternative to the short-period requirements currently in MIL-F-8785B. Although the exact numerical boundaries have been challenged (see e.g. Reference 5), the current requirements are reasonably well supported for classical configurations, in which the short-term response is governed by a single pair of complex poles. A frequent product of advanced control systems is a higher-order dynamics system such that there is no unique "short-period" mode. The proposed revision uses a second-order transfer function to define an equivalent short-period mode and a corresponding first-order numerator with an added pure time delay for pitch response. The proposed boundaries were derived from the Neal-Smith criterion of Reference 6. Reference 1 suggested another form of the Neal-Smith criterion for consideration. As-yet-unpublished correlation work in AFFDL/FGC indicates problems with this alternate criterion. A better alternative may be stated in terms of the required bandwidth and allowable peak overshoot and droop - the closed-loop parameters which Neal and Smith suggested that pilots use to adjust their dynamic response (Reference 6). Additional data for other Flight Phases and

other airplane Classes, and for Level 3 generally is needed.

3.2.2.2 Control feel and stability in maneuvering flight at constant speed: This proposal would remove the requirements on control surface deflection, provided that the pilot control force and deflection are stable. This allows an unstable airframe, with stability provided through an augmentation system, although the consideration of a Special Failure State (3.1.6.2.1) still applies. Control margin then becomes a safety consideration; see the proposed paragraph 3.4.11.

3.2.2.2.1 Control forces in maneuvering flight: The proposed change defines a range over which linearity is expected. As pointed out in a comment, a limit of 3 g would be inappropriate at lower speeds. We recommend, therefore, that the upper limit for Class IV be made the same as the other Classes. We feel that in arriving at this suggested change we have given consideration to opposing points of view regarding control-force linearity.

For Level 3, the present 3 lb per g limit would seem to serve the B-1 better than the proposed 2 lb per g.

3.2.2.2.1.1 Sidestick controller forces: This proposed paragraph does little more than recognize the existence of sidestick controllers, because there is insufficient data to support definitive requirements. AFFDL/FGC is currently assembling such a data base; results to date are presented elsewhere in this report. (Interestingly one comment supports having only qualitative requirements.)

3.2.2.2.2. Control Motions in maneuvering flight: A minimum force per deflection gradient is added for sidestick controllers.

3.2.2.3 Compatibility of steady maneuvering forces and pitch sensitivity: This proposed new paragraph repeats the Control Anticipation Parameter (CAP) or $\omega^2 n_{sp}/n_{\alpha}$ values of the current short-period requirements (3.2.2.1.1). The application here is to aircraft response to control input (independent of the short period requirements). As stated in Reference 1, this requirement is added here so as to apply where the proposed 3.2.2.1.3 is used in place of the current short-period requirements.

3.2.2.4 <u>Longitudinal pilot-induced oscillations</u> through 3.2.2.4.3 <u>Control system phase lag</u>: These expanded PIO requirements are from work by Ralph Smith (Reference 7). Comments received to date indicate acceptance qualified by lack of both prior knowledge of, and experience with, the requirements.

An effort is currently underway in AFFDL/FGC to verify both the theory and its use for design requirements. If certain pre-conditions are satisfied, the final criterion of Smith's theory requires that the phase margin of the normal acceleration at the pilot station to stick force transfer function (with pilot delay of 0.25 secs) should be positive to preclude PIO. Figure 4 shows the mimimum phase of the above transfer function versus PIO pilot rating for the LAHOS data (presented elsewhere in this report). The trend is obviously correct, however a first impression is that the criterion may be overly conservative. This validation work is continuing, and the results will be published as soon as appropriate.

3.2.3.5 <u>Longitudinal control forces in dives - Service Flight Envelope</u>: A minor change to make control forces for one-handed wheel operation the same as for center-stick operation.

II.5 Lateral-directional requirements

We still do not know enough to write a good Dutch roll requirement. It is generally a nuisance mode; perhaps delineating what is not satisfactory, acceptable, or safe is more difficult than stating what is needed. Further, the typical roll and Dutch roll modes both have significant short-term responses - thus a larger number of parameters must be considered together than for longitudinal motion. The present requirements on lateral-directional dynamic response characteristics, though overly complex, provide only fair boundaries even for classical types of response. Suggestions for improvement have tended to increase complexity rather than to simplify. So for now we do what we feel we can do to upgrade the existing requirements.

For all the dynamic requirements, the data base consists almost entirely of configurations which have characteristics which do not change drastically from controls-free to controls-fixed. With modern flight control systems this is no longer the only viable design alternative, and one might ask what consequences such changes in dynamics hold for flying qualities requirements (both longitudinal and lateral-directional).

3.3.1.1 Dutch roll: The numerical Dutch roll changes are those recommended by Calspan (Reference 2). Also, by deleting the requirement for control-surface-fixed Dutch roll stability we eliminate a roadblock to relaxed static stability. The possibility of increased stability augmentation reliability, we feel, warrants at least allowing such measures to be considered in design trade-offs. The same caveats apply as in the longitudinal case.

3.3.1.3 Spiral stability: The changes to spiral stability requirements also are the Calspan recommendations (Reference 2).

3.3.1.4 Coupled Roll-spiral oscillations: The coupled roll-spiral oscillation boundaries are based on the data presented in Reference 2, but it was felt that lines of constant better fit that data. As pointed out in the proposed change material, some autopilot and low-maneuverability control wheel steering modes inherently produce roll-spiral coupling in order to improve stabilization, control and ease of piloting.

II.6 Roll Control Requirements

There have been a number of complaints that the roll response requirements of MIL-F-8785B are too stringent at the extremes of the flight envelopes. Furthermore, there has also been a reluctance to accept literally that those roll requirements apply throughout the applicable load factor range. There is also some confusion caused by an apparent conflict with the flight loads specification (MIL-F-8861) which only requires full control 360° rolls at one g load factor. The revisions proposed are in response to the above concerns.

The references to "$V_{o_{min}}$" and "V_{min}" in the speed range portions of the proposed revisions should be changed to "V at $n_o(+)$" and "V at n(+)" to agree with the definitions of section 6. That is, the speed ranges are intended to exist at all applicable normal load factors as functions of the speed-load factor boundaries at a given altitude.

3.3.4 Roll control effectiveness: The proposed format modification leaves only Classes I and II in this paragraph. The specification values are unchanged from MIL-F-8785B, primarily because of a lack of data, although one reviewer suggests that the Class II requirements are too stringent.

3.3.4.1 Roll performance for Class IV airplanes through 3.3.4.1.2 Roll performance in Flight Phase GA: The Class IV requirements are the most extensively modified, but the change is not as massive as it would seem from the increase in the size of the section. The requirements are adjusted to relax the requirements at the speed extremes and the bank angle change has been modified to be compatible with the speed at which the roll performance will be demonstrated. Maybe more importantly, the requirements are clearly applied with respect to aircraft load factor. The roll performance for 360 degree rolls are proposed to apply only at one g which, because it agrees with the current requirement in the loads specification, can be tested without special planning in a typical test program. The roll performance requirements at elevated load factor are proposed as bank-to-bank rolls through bank angles of 180 degrees or less. These rolls will be initiated at load factors up to the maximum for the applicable flight envelope. No change has been proposed for the roll-pitch-yaw coupling paragraph (3.4.4) though this is an important interface with the structural laods functional area which will be addressed in future work.

3.3.4.2 Roll performance for Class III airplanes: The Class III requirements also have been adjusted to relax the requirements at the speed extremes and to make the bank angle change compatible with the speed at which the roll performance would be demonstrated.

3.3.4.5 (Old) Rudder-pedal-induced rolls: The proposed revision eliminates the roll-rate-due-to-rudder requirement. This was done

as a result of pilot comments which indicated that some tasks were easier if the aircraft had no roll due to sideslip. The elimination led to a comment that perhaps the requirement should be retained to apply only to single-seat aircraft. In any event, little data exists to support any requirement and the subject of whether to limit the amount of roll due to rudder or sideslip has not even been addressed.

II.7 Control System Requirements

3.4.11 Control margin: This new paragraph is proposed mainly to complement other changes. With an unstable airframe now allowed, we must ensure that control authority and rate are sufficient to provide stability under all reasonable conditions and also to recover from almost any upset and departures from controlled flight. Comments show general acceptance of the proposal.

3.5.3 Dynamic characteristics: The current 3.5.3 and 3.5.3.1 are restated in the proposal, as requirements on the relation between airplane response and controller inputs. One negative comment received was to delete the "guide for airplanes with simple, conventional control systems." The "guide," retained from MIL-F-8785B Table XIII, was added back into the 1977 version of Reference 1 at the suggestion of the Navy based on their experience. In addition, the caveat on "simple conventional control systems" may be unnecessary.

3.5.5 Failures: A restatement of the same requirements, plus explicit mention of configuration change as a way of adapting to a failure situation.

3.5.5.1 Failure transients: This proposal changes the requirements so that, for Level 1 conditions after the failure, the allowable transients are the same as for Level 2. Additional limits are also stated for bank angle and vertical and lateral translations. For the Level 1 condition the requirements are far less severe. Comments on the proposal include a suggestion for limits on angle of attack and sideslip, and a suggestion for less severe requirements for Level 2 conditions.

3.5.6 Transfer to alternate control modes: A restatement of the

objective without specific "how-to" requirements.

3.5.6.1 Transfer transients: The proposal is for less severe requirements, consistent with the changes to 3.5.5.1.

3.6.1 Trim system: A minor clarification - the Level 3 requirement applies to steady-state not transient forces.

3.6.1.2 Rate of trim operation: The proposed change directs center-stick control force limits for one-handed wheel operation.

3.6.3 Transients and trim changes: Minor changes for clarification.

II.8 Atmospheric Disturbance Model

The initial approach was to use the disturbance model in MIL-F-8785B and revise it only where necessary. A major deficiency of that model is its characteristics at low altitudes. The proposed solution is to provide a model specifically for low altitudes based on the assumption that the terminal Flight Phases are separated from other Flight Phases in simulation, in analyses and also in flight test, to large degree. It is believed that the changes made (with the exception of terminology changes) were necessary to upgrade the existing model while still retaining the philosophy of presenting minimum acceptable requirements. More sophisticated (complex) disturbance models are available, e.g. References 8 and 9, and their use is certainly not discouraged.

A second change is to give three values of the disturbance intensities, LIGHT, MODERATE and SEVERE. These terms correspond to specific probabilities to be used in defining explicitly (together with the proposed 1.5) the effects of disturbances on flying qualities. In general, the proposed changes are consistent with the changes to 1.5 and the previous discussion applies here also. General acceptance of all the proposed disturbance model may depend on acceptance of 1.5.

3.7.1 Use of environmental models: The change to this paragraph introduces the use of a model specifically for low altitudes. It also dictates consideration of changes in altitude for the low altitude Flight Phases. In the following changes the original model in MIL-F-8785B (3.7.2 through 3.7.4.2) is basically the medium/high altitude model of the revision (3.7.2 through 3.7.3.4).

3.7.2.3 Discrete gust model: The "1 - cosine" gust of MIL-F-8785B has been retained, although half a period is specified. This makes the gust model more general. With the right parameters, for instance, the discrete gust can represent a simple wind shear. In a different application, a pair of gusts can be used with the spacing chosen to produce the largest disturbance. This approach is illustrated in Figure 5, and is based on the work of Jones of the RAE. The remaining deficiency appears to be a lack of quantitative requirements for the allowable responses (discussed in Section III.4, Vol. II, Reference 1).

3.7.3.2 Turbulence intensities: The turbulence intensities proposed for the three probabilities are consistent with other models, such as MIL-F-9490D. The proposed variation with altitude, however, is an obvious simplification. With the possibility of wide variation in local conditions, the use of complex curves for global average does not seem justified. According to Reference 10, the probabilities tabulated in MIL-F-9490D were calculated using MIL-A-008861A values of proportions of flight time in nonstorm and storm turbulence at a given altitude, and also the expected values of intensity in nonstorm and storm turbulence at that altitude.

Several questions might be raised. One is nomenclature, equating intensity to probability: for example in Figure 4, Reference 1, LIGHT turbulence does not exist above 30,000 ft. Related is the alternative of stating intensity per se. Reference 2 points out that the expected value of nonstorm turbulence, once it is encountered, is relatively invariant with altitude. It is the probability of finding turbulence at all that varies with altitude. For flying qualities evaluation they recommend this value, around 2.3 ft/sec as "light-to-moderate turbulence which [the pilot] is likely to encounter," plus more severe turbulence. Then the proper values of either intensity or probability must be chosen.

3.7.3.4 Gust magnitudes: A major inclusion in the proposal is severe discrete gust values to be consistent with structural requirements. This change is intended to ensure that control can be maintained even in large disturbances. It would be impractical to show compliance with this requirement in flight test, and probably also in a simulator. It is suggested that compliance be shown analytically.

3.7.4 Low altitude environmental model: This paragraph introduces the requirement to consider the cumulative effects of wind shear, turbulence and gusts - defined in the following paragraphs. Reference is made to the use of non-Gaussian turbulence models. In particular, Jones' Discrete Gust Spectrum model is much better developed now than it was when Reference 1 was first drafted in 1975.

3.7.4.2 Wind shear: A simple logarithmic profile is specified to provide a natural variation of wind speed with altitude. Either a discrete gust or vector shear can be used to evaluate the effects of the more unusual or extreme wind shears, in the absence of more definitive models.

3.7.4.3 Vector shear: This paragraph was added to produce variations representative of the more unusual wind conditions such as may be produced by frontal conditions, temperature inversions, thunderstorm gust fronts, etc.

3.7.5 Application of the environmental models in analyses: For the medium/high altitudes this paragraph is unchanged from MIL-F-8785B. The proposed change is to require the turbulence axes to be referred to the mean wind direction at low altitudes. This is consistent with the definitions of measured turbulence velocity components. The angular velocity perturbations are retained in aircraft body axes, however, because the expressions used are simple approximations.

III. CONCLUDING REMARKS

This paper summarizes revisions proposed for MIL-F-8785B, "Military Specification, Flying Qualities of Piloted Airplanes." Additional justification has been added to answer comments already received. It is

intended to stimulate discussion and hopefully it will help to finalize industry coordination of the proposed revisions. With industry and government agreement, Amendment 3 to MIL-F-8785B will be drafted as soon as possible.

IV. REFERENCES

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9. Jones, J.G., "Turbulence Models for the Assessment of Handling Qualities during Take-off and Landing," AGARD Meeting on <u>Handling Qualities Criteria</u>, AGARD-CP-106, June 1972.

10. Townsend, J.L. and Raymond, E.T., "Background Information and User Guide for MIL-F-9490D," AFFDL-TR-74-116

1.1	Scope - clarification
1.2	Application - guidance on add'l rqmts; mentions 9490 & 18244
1.5	Levels - while the Levels must apply to the flight vehicle being procured, there is recognition that atmospheric disturbances themselves degrade flying qualities.
3.1.1	Operational Missions - clarification
3.1.3	Moments and Products of inertia - I_{xz} added
3.1.9	Permissible Flight Envelope - left for contractor to define
3.1.10	Appl. of Levels - accounts for atm. disturbances in 1.5
3.2.1.1	Longitudinal static stability - allows a degree of divergence for Level 3
3.2.2.1.3	(New Higher order systems - an equivalent classical systems approach to short-period dynamics when FCS adds roots, or an alternative Calspan formulation of the Neal-Smith criteria
3.2.2.2	Maneuvering flight - allows basic airframe instability at procuring activity's discretion (3.1.6.2.1, Special Failure States); rqmts refined, side stick recognized.
3.2.2.4	Longitudinal pilot-induced oscillations - Ralph Smith's criteria
3.2.2.5	Longitudinal force in dives - refined
3.3.1.1	Dutch roll - refined; surface-fixed stability rqmt deleted
3.3.1.3	Spiral stability - refined
3.3.1.4	Coupled roll - spiral oscillation - allowed, within $\zeta\omega$ limits
3.3.4	Roll control - reorganized, refined; rqmts lessened near flt envelope boundaries
3.4.11	(new) Control margin - controllability of "control-configured vehicles"
3.5.3	Dynamic characteristics - refined; the specified values of lag are not always vlaid
3.5.5	Failures - recast; ± .5g transient (instead of ± .05g) if Level 1 is retained after the failure
3.5.6	Transfer to alternate control modes - recast; 0.1g transient (instead of 0.05g) allowed in Operational Flt Envelope
3.6.1	Trim system - Refined
3.6.2	Speed and flight-path control devices - clarified
3.6.3	Transients and trim changes - adds thrust reverser; refined
3.7	Atmospheric disturbances - refined discrete gust model, using maxim from structural spec; "light,""moderate," "severe" intensities defined; wind logarithmic shear & vector shear added to low altitude model
6.1	Intended use - refined

Throughout - change "elevator, aileron, rudder" to "pitch, roll, yaw controls"

Figure 1. Summary of Proposed MIL-F-8785B Revisions

COOPER-HARPER RATING*		ATMOSPHERIC DISTURBANCES		
		LIGHT	MODERATE	SEVERE
1-3.5	Level 1	Flying qualities clearly adequate for the mission Flight Phase	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. Category A Flight Phase can be terminated safely, and Category B and C Flight Phases can be completed.
$\leq$ 6.5	Level 2	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed	Flying qualities such that control can be maintained long enough to fly out of the disturbance.
$\leq$ 9	Level 3	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.	Flying qualities such that control can be maintained long enough to fly out of the disturbance.	No requirement
$\leq$ 10				

*** If Pilot does not mentally compensate for effects of turbulence, etc.**

Figure 2 Proposed Levels of Flying Qualities with Atmospheric Disturbance Effects

ATMOSPHERIC DISTURBANCES

LEVEL	LIGHT	MODERATE	SEVERE
1	Flying qualities clearly adequate for the mission Flight Phase	Flying qualities clearly adequate for the mission Flight Phase	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists
2	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed
3	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed	Flying qualities such that control can be maintained long enough to fly out of the disturbance

Figure 3. Possible Flying Qualities Levels for 'All-Weather' Aircraft

Solid symbols may be pitch sensitivity not PIO.

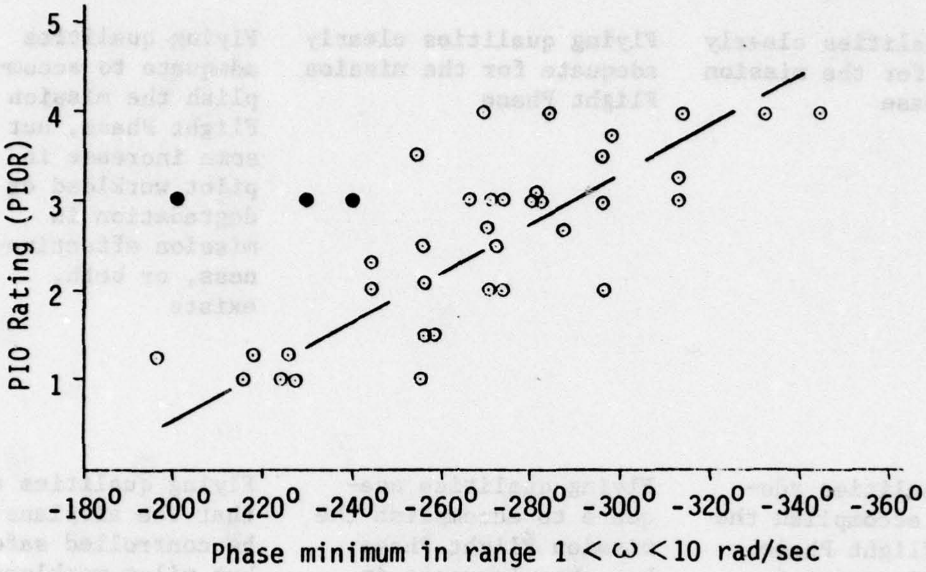
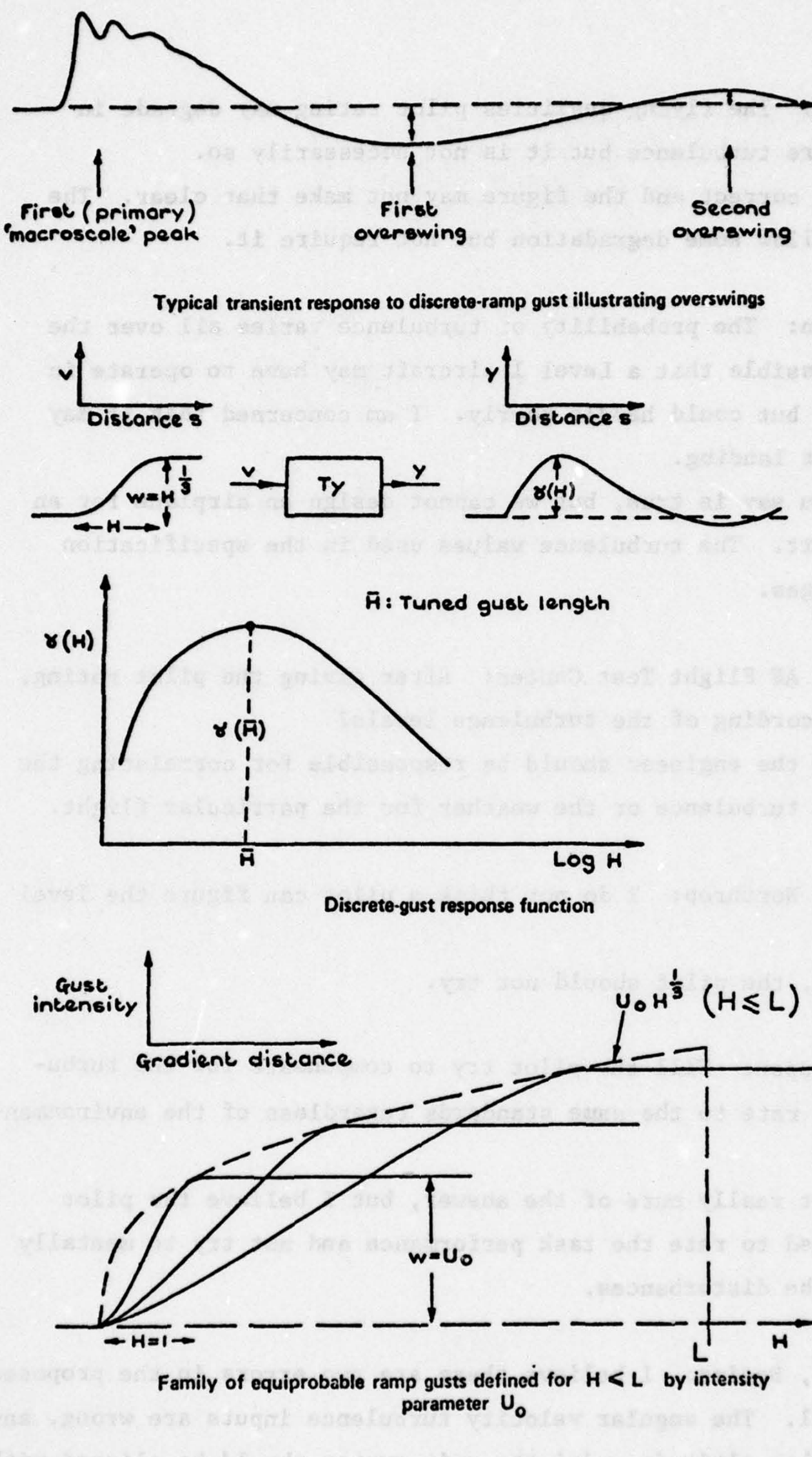


Figure 4. Correlation of $\frac{a_{z_p}}{F_s}$ phase margin with PIO rating.



Typical transient response to discrete-ramp gust illustrating overswings

Discrete-gust response function

Family of equiprobable ramp gusts defined for $H < L$ by intensity parameter U_0

Figure 5. Discrete gust application of Jones

Don West, Boeing: The flying qualities pilot rating may degrade in moderate or severe turbulence but it is not necessarily so.

Answer: That is correct and the figure may not make that clear. The proposal is to allow some degradation but not require it.

Bud Iles, Grumman: The probability of turbulence varies all over the world. It is possible that a Level 1 aircraft may have to operate in a turbulent area but could handle poorly. I am concerned that it may be a problem just landing.

Answer: What you say is true, but we cannot design an airplane for an individual airport. The turbulence values used in the specification are global averages.

Capt Roy Martin, AF Flight Test Center: After giving the pilot rating, who makes the recording of the turbulence levels?

Answer: I think the engineer should be responsible for correlating the results with the turbulence or the weather for the particular flight.

Jerry Lockenour, Northrop: I do not think a pilot can figure the level of turbulence.

Answer: I agree, the pilot should not try.

Chick Chalk, Calspan: Will the pilot try to compensate for the turbulence or will he rate to the same standards regardless of the environmental conditions?

Answer: I am not really sure of the answer, but I believe the pilot should be required to rate the task performance and not try to mentally compensate for the disturbances.

Dwight Schaeffer, Boeing: I believe there are two errors in the proposed disturbance model. The angular velocity turbulence inputs are wrong, and second, for the low altitude model the axis system should be aligned with the relative wind vector not the actual wind vector.

Answer: I agree that the angular velocity turbulence expressions are crude approximations in lieu of something better; this is mentioned in the revision document. I am not sure I agree with the second point but it will be checked.

Bud Iles, Grumman: I have some questions about the proposed disturbance model. Are gusts in constant directions? Are low altitude gusts in the same direction? What are the gusts for severe turbulence? I think the requirements may be too lenient.

Answer: The turbulence model provides gust inputs in the three axes; the relation between these inputs is a function of altitude close to the ground. Also included at low altitude are winds, wind shear and changes in wind direction as functions of altitude. The severe disturbances are equivalent to typical thunderstorn intensities, so I do not think it is lenient.

John Schuler, Boeing: In severe turbulence aren't we relaxing the requirements so that the designer will always provide the minimum, i.e., a pilot rating of 6.5 in moderate disturbances?

Answer: In a sense the specification provides only minimum requirements. This does not prevent better flying qualities being designed in, for instance a gust alleviation system could still be used if justified for a particular design.

Chick Chalk, Calspan: What if the aircraft must have clearly adequate flying qualities in the moderate or severe disturbances.

Answer: The procuring activity will always have the option of increasing the requirements, either by lowering the probabilities associated with the moderate and severe disturbances or by altering the progression of allowable degradation from light to moderate to severe disturbances.

NORTHROP REVIEW OF MIL-F-8785B

PROPOSED REVISIONS

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ACKNOWLEDGEMENT

The author would like to acknowledge the many Northrop contributors to this effort. Their individual efforts provided the major working base from which this paper was derived. These contributions were made by Controls Technology, F-5 Aerosciences, and Advanced Design. Although the contributors were many the author would especially like to acknowledge W. E. Nelson, Jr, W. J. Gaugh, and R. N. Kandalaft for their major contributions and for their help in the review of this paper.

INTRODUCTION

The AFFDL package of proposed revisions to MIL-F-8785B, the Military Specification for Flying Qualities of Piloted Aircraft (Ref. 1), was received by Northrop's Aircraft Group in mid-March 1978. A review was requested with comments to be returned to the government at a symposium and workshop to be conducted in September. The revision package was distributed to appropriate organizations within Northrop. As a result of the initial review, certain topics were identified for which Northrop had as yet unpublished data which would be valuable to the specification activity. Figure 1 shows those paragraphs for which specific contributions have been made. This paper gives a summary of each area along with example data to illustrate the nature of the results.

LEVELS OF FLYING QUALITIES (1.5)

The current version of MIL-F-8785B (Ref. 2) defines levels of flying qualities using one paragraph descriptions which are consistent with levels as defined by the Cooper-Harper rating scale (Ref. 3). Although it is not explicitly stated in the specification, Level 1 corresponds to pilot ratings (PR) less than 3.5, Level 2 to PR's between 3.5 and 6.5, and Level 3 to PR's between 6.5 and 9.5. See Figure 2.

• 1.0	SCOPE AND CLARIFICATION	3.3.1	LAT.-DIR. DYNAMICS
3.1	GENERAL REQUIREMENTS	• 3.3.4	ROLL CONTROL EFFECTIVENESS
• 3.2.1	LONG. STAB. WITH RESPECT TO SPEED	3.4.11	CONTROL MARGIN
• 3.2.2.1	LONG. SHORT PERIOD	3.5	CHARACTERISTICS OF PRIMARY FCS
• 3.2.2.2	CONTROL FEEL AND STAB. IN MANEUVERING	3.6	CHARACTERISTICS OF SECONDARY FCS
• 3.2.2.3	COMPATIBILITY OF MAN. FORCES AND PITCH SENS.	• 3.7	ATMOSPHERIC DISTURBANCES
• 3.2.2.4	PIO	6.1	INTENDED USE

NOTE: • SYMBOL IDENTIFIES THOSE PARAGRAPHS REVIEWED

FIGURE 1. PARAGRAPHS REVIEWED IN DETAIL BY NORTHROP

	8785B DEFINITIONS		COOPER HARPER RATINGS
LEVEL 1	FLYING QUALITIES CLEARLY ADEQUATE FOR THE MISSION FLIGHT PHASE	=	< 3.5
LEVEL 2	FLYING QUALITIES ADEQUATE TO ACCOMPLISH THE MISSION FLIGHT PHASE, BUT SOME INCREASE IN PILOT WORKLOAD OR DEGRADATION IN MISSION EFFECTIVENESS, OR BOTH, EXISTS	=	3.5 – 6.5
LEVEL 3	FLYING QUALITIES SUCH THAT THE AIRPLANE CAN BE CONTROLLED SAFELY, BUT PILOT WORKLOAD IS EXCESSIVE OR MISSION EFFECTIVENESS IS INADEQUATE, OR BOTH. CATEGORY A FLIGHT PHASES CAN BE TERMINATED SAFELY, AND CATEGORY B AND C FLIGHT PHASES CAN BE COMPLETED	=	6.5 – 9.5

FIGURE 2. CURRENT SPECIFICATION DEFINITION OF LEVELS OF FLYING QUALITIES

The proposed revision modifies the definition of Levels as shown by Figure 3. The objective is to explicitly recognize the effect of atmospheric disturbances on flying qualities. Sufficient data are available to show that aircraft control and/or pilot workload can be significantly degraded while operating in moderate to severe environments (Ref. 4, etc.). It is recommended however that this should not change the definition of Levels. The definition of Levels corresponding to the Cooper-Harper Scale should be maintained. Figure 4 shows that there are three primary factors which influence a flying qualities evaluation. These are the system (the airplane in a normal or failure state condition, the flight condition (where in the flight envelope), and the environmental definition (atmospheric condition). When these three factors are defined and a task evaluation is conducted a certain task performance and pilot workload results which correspond to a pilot rating and therefore a Level.

	ATMOSPHERIC DISTURBANCES		
	LIGHT	MODERATE	SEVERE
LEVEL 1	COOPER-HARPER LEVEL 1 DEFINITION	COOPER-HARPER LEVEL 2 DEFINITION	COOPER-HARPER LEVEL 3 DEFINITION
LEVEL 2	COOPER-HARPER LEVEL 2 DEFINITION	COOPER-HARPER LEVEL 3 DEFINITION	COOPER-HARPER LEVEL 3 WITH NEW TASK DEF
LEVEL 3	COOPER-HARPER LEVEL 3 DEFINITION	COOPER-HARPER LEVEL 3 WITH NEW TASK DEF	NO REQUIREMENT

FIGURE 3. PROPOSED REVISION TO THE DEFINITION OF LEVELS OF FLYING QUALITIES

MIL-F-8785B recognizes the effect of failures and flight condition on flying qualities in paragraph 3.1.10 of the specification by allowing degraded flying qualities for Failure Status and for operation outside the Operational Envelope. It is recommended that a better way to accommodate the effect of atmospheric disturbances is to include it in paragraph 3.1.10 as part of a three dimensional space which calls out the required Level of flying qualities as a function of the three primary factors that influence flying qualities. Figure 5 shows the form of this three dimensional space. The Levels assigned to each element of this space are those taken from the specification and the proposed revisions. This framework also makes it clear that the procuring activity can change the required Level of flying qualities for a specific

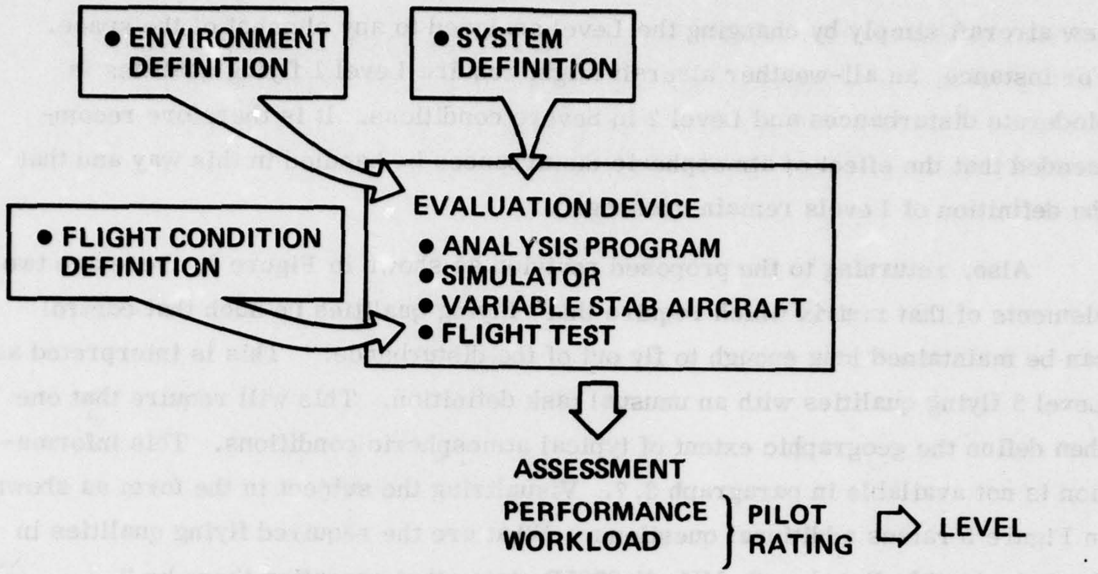

FIGURE 4. THREE FACTORS THAT INFLUENCE THE FLYING QUALITIES ASSESSMENT OF AN AIRCRAFT

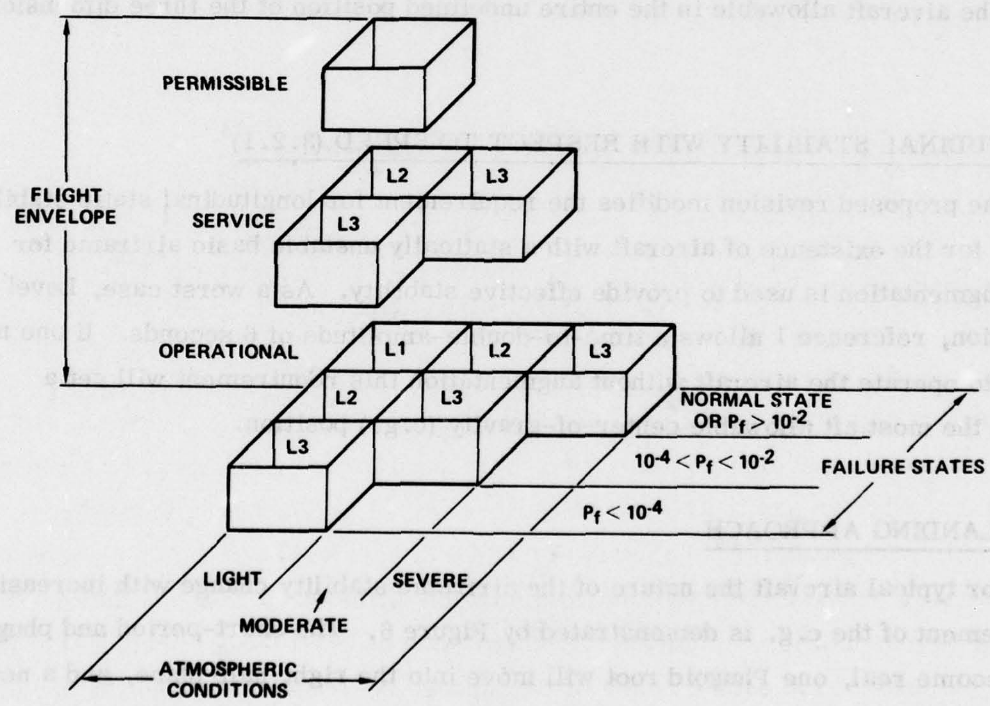

FIGURE 5. ALLOWABLE DEGRADATIONS IN THE LEVEL OF FLYING QUALITIES

new aircraft simply by changing the Level assigned to any element of the space. For instance, an all-weather aircraft might require Level 1 flying qualities in Moderate disturbances and Level 2 in Severe conditions. It is therefore recommended that the effect of atmospheric disturbances be handled in this way and that the definition of Levels remain unchanged.

Also, returning to the proposed revision as shown in Figure 3, there are two elements of that matrix which require that "Flying qualities be such that control can be maintained long enough to fly out of the disturbance." This is interpreted as Level 3 flying qualities with an unusual task definition. This will require that one then define the geographic extent of typical atmospheric conditions. This information is not available in paragraph 3.7. Visualizing the subject in the form as shown in Figure 5 raises additional questions. What are the required flying qualities in the Permissible Envelope? MIL-F-8785B states that operation there be "........ allowable and possible." Does that mean Level 3? Also what about the remainder of this three dimensional space of Figure 5? Figure 3 states "no requirement" for Level 3 in severe atmospheric conditions. Does this mean that loss of the aircraft is allowable under these conditions? And, when viewed in the form of Figure 5, is loss of the aircraft allowable in the entire undefined position of the three dimensional space?

LONGITUDINAL STABILITY WITH RESPECT TO SPEED (3.2.1)

The proposed revision modifies the requirement for longitudinal static stability to allow for the existence of aircraft with a statically unstable basic airframe for which augmentation is used to provide effective stability. As a worst case, Level 3 condition, reference 1 allows a time-to-double-amplitude of 6 seconds. If one must be able to operate the aircraft without augmentation this requirement will set a limit on the most aft allowable center-of-gravity (c.g.) position.

YF-17 LANDING APPROACH

For typical aircraft the nature of the airframe stability change with increasing aft movement of the c.g. is demonstrated by Figure 6. The short-period and phugoid roots become real, one Phugoid root will move into the right-half-plane, and a new "third mode" will form. For the Northrop YF-17 the sequence of events was first the short period roots would become real. The Phugoid roots would become real,

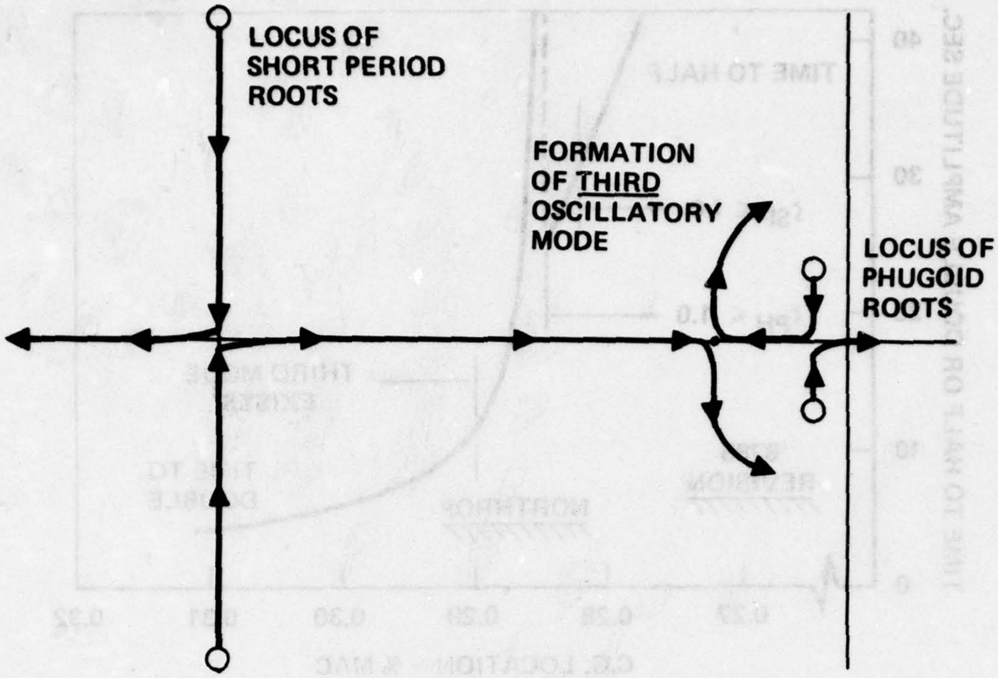

FIGURE 6. TYPICAL MIGRATION OF THE ROOTS OF THE LONGITUDINAL CHARACTERISTIC EQUATION WITH INCREASING AFT MOVEMENT OF THE CENTER OF GRAVITY

the one root would become unstable, then the third mode would form. Figure 7 shows the effect of c.g. location on the longitudinal stability of the YF-17 in the landing approach phase. The aircraft is stable for c.g. positions forward of about 28.5% mac and statically unstable for more aft c.g.'s. Based upon extensive piloted flight simulations, Northrop established a minimum safe time-to-double-amplitude of 4 seconds. The proposed specification revision of 6 seconds is considered to be in agreement with the Northrop value. The 6 second number is based upon in-flight simulation where the additional task realism normally results in more conservatism.

It is interesting to note the sensitivity of the modal changes with c.g. position. With a one percent change in c.g., static stability can change significantly. It has long been known that the sensitivity of time-to-double-amplitude in the pitch axis is largely determined by the stability derivative C_{m_u} (or M_u). See Figure 8. This parameter is currently paid little attention in airplane design and is very difficult to predict accurately.

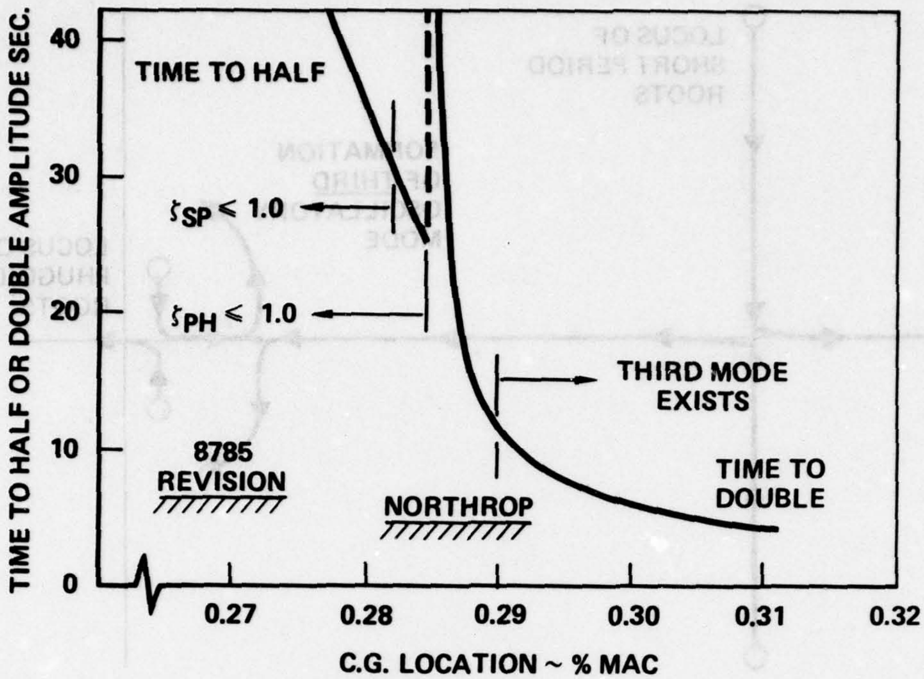

FIGURE 7. VARIATION IN YF-17 LONGITUDINAL STABILITY WITH CENTER OF GRAVITY POSITION FOR LANDING APPROACH

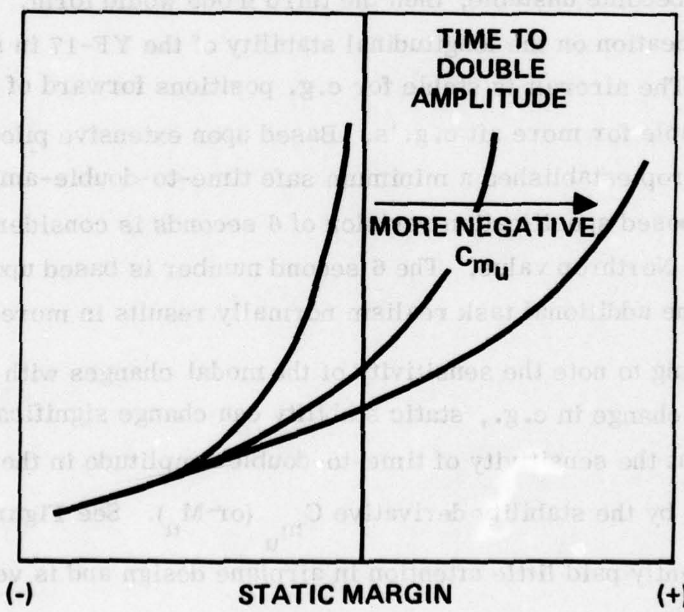

FIGURE 8. THE STABILITY DERIVATIVE, CHANGE IN PITCHING MOMENT WITH SPEED, DETERMINES THE SENSITIVITY OF STATIC STABILITY TO C. G. POSITION

YF-17 TRANSONIC

The effect of failure transients and control following the failure was extensively tested in the YF-17 program. "Failure Modes and Effects" piloted simulations were conducted in which selected critical failures were simulated in both typical flight conditions and in conditions chosen to be most critical for each particular Failure State. For longitudinal control the Pitch Control Augmentation System (PCAS) failure was most critical at aft c.g.'s. For the YF-17 at 0.85 M the elastic neutral point is at 29% of the mean aerodynamic chord (mac) and the elastic maneuver point is at 32% mac. Figures 9 and 10 are typical data from those simulation tests. Transient and peak load factor values are shown for each test condition as the c.g. is moved further and further aft. Pilot ratings were also taken for each task which was to maintain control during and after the failure. Based upon the composite of these simulation tests, the flying qualities correlated most readily with maneuver margin and stick force gradient as shown by the solid lines on Figure 11. These simulation trends were verified by a limited number of flight test points. These data indicate that a minimum maneuver margin of about 1.5% mac and a minimum stick force gradient of about 2 lbs/g are reasonable Level 3 design conditions.

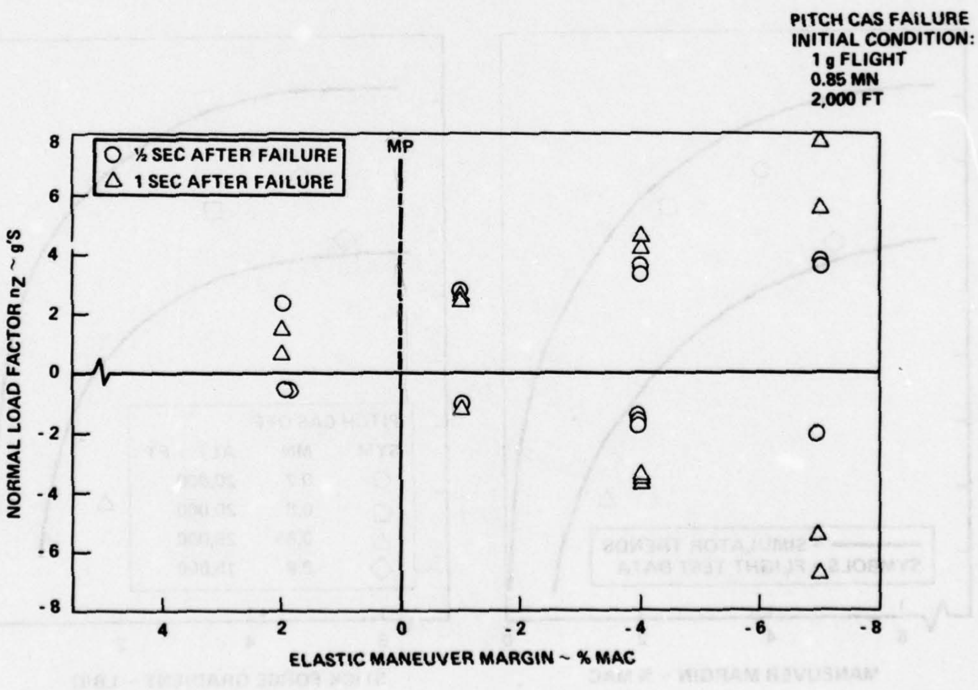

FIGURE 9. EFFECT OF MANEUVER MARGIN ON NORMAL LOAD FACTOR TRANSIENTS FOLLOWING A PITCH CAS FAILURE

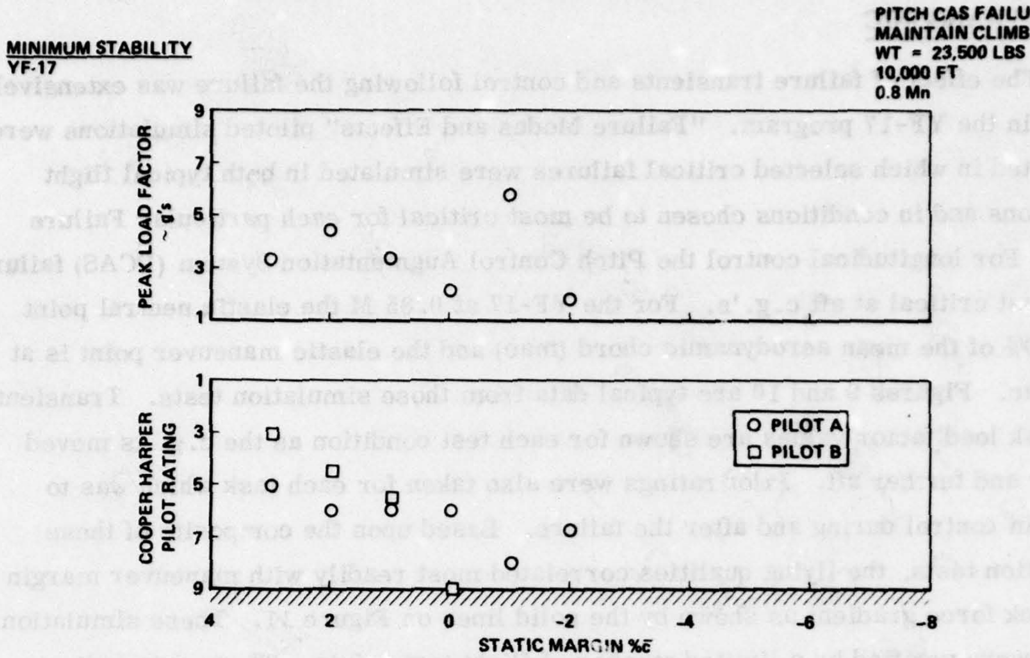

FIGURE 10. EFFECT OF STATIC MARGIN ON LONGITUDINAL CONTROL FOLLOWING A PITCH CAS FAILURE

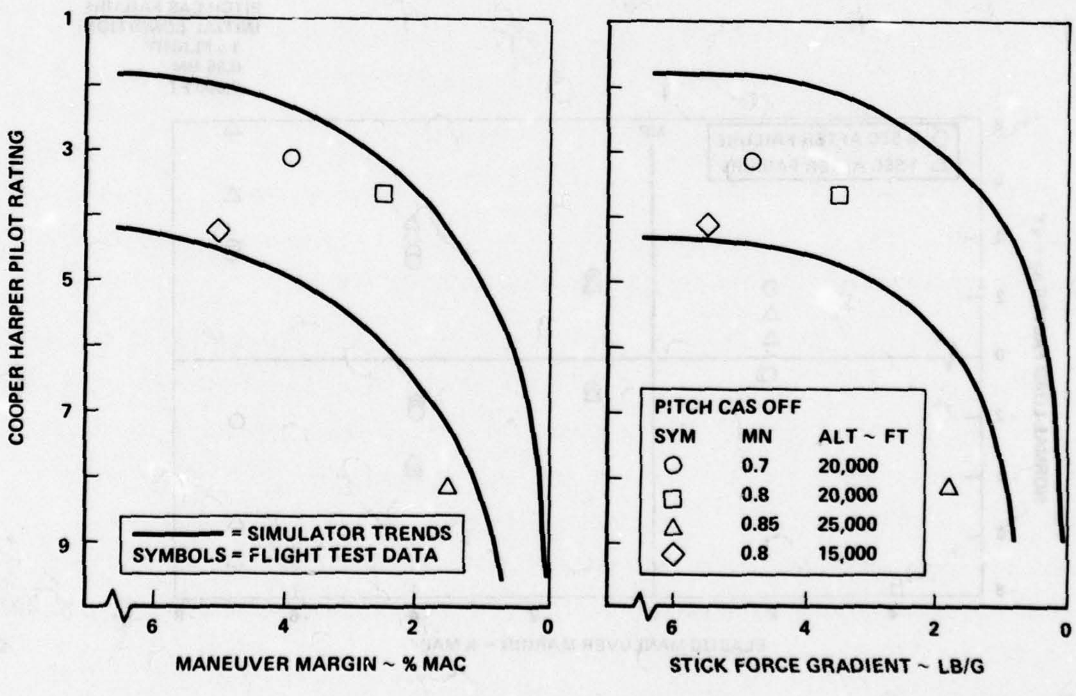

FIGURE 11. FLYING QUALITIES CORRELATE WITH MANEUVER MARGIN AND STICK FORCE GRADIENT

LONGITUDINAL SHORT PERIOD RESPONSE (3.2.2.1)

One of the primary motivations for revising 8785B at this time is to insure that it will accommodate highly augmented aircraft for which the resulting dynamics are far removed from those of the basic airframe. Figure 12 lists the typical problems that occur when attempting to compare such aircraft with the current specification. MIL-F-8785B longitudinal short period response requirements are now stated in terms of classical modal parameters which are difficult to identify for highly augmented aircraft.

LARGE ORDER SYSTEM

- MANY ROOTS ON ROOT LOCUS
- MANY TERMS IN TRANSFER FUNCTION

NON-LINEAR SYSTEM

- PROBLEM FOR ANY LINEAR METHOD

NON-CLASSICAL MODES

- LIMITED SYSTEMATIC DATA BASE

FIGURE 12. HIGHLY AUGMENTED AIRCRAFT PRESENT MANY PROBLEMS FOR THE CURRENT SPECIFICATION CRITERIA

Northrop is currently using a variety of alternate criteria as design guides for highly augmented aircraft. Figure 13 lists those approaches which are currently being used. Northrop's experience and methodologies for pilot-in-the-loop analyses are well documented in a recent contract publication, reference 5.

Figure 14 shows the pitch response criteria to which the YF-17 was designed. The total augmented frequency response of pitch rate to stick force was required to fall within the shaded region. These bounds are based primarily upon the Neal-Smith data base of reference 6. This criteria can be used regardless of the order of the subject airplane system. Non-linear effects can be accommodated if one tests for the input amplitude dependent nature of the frequency response. Unacceptable airplanes are most conspicuous on this criteria by the increased phase lag at moderate frequencies as shown in Figure 15.

APPROACHES

- FREQUENCY DOMAIN (BODE) BOUNDARIES
- TIME RESPONSE BOUNDARIES
- EQUIVALENT SYSTEM PARAMETERS
- PILOT-IN-THE-LOOP ANALYSIS

FIGURE 13. APPROACHES TO FLYING QUALITIES CRITERIA FOR HIGHLY AUGMENTED AIRCRAFT

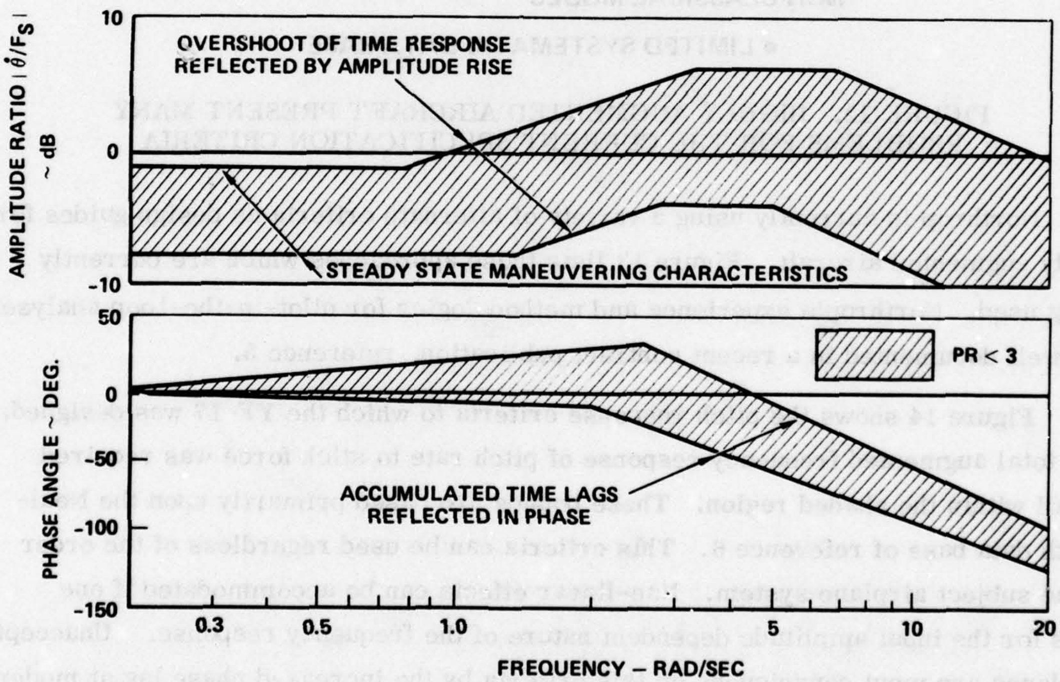

FIGURE 14. LEVEL 1 FREQUENCY RESPONSE REGION FOR LONGITUDINAL CONTROL

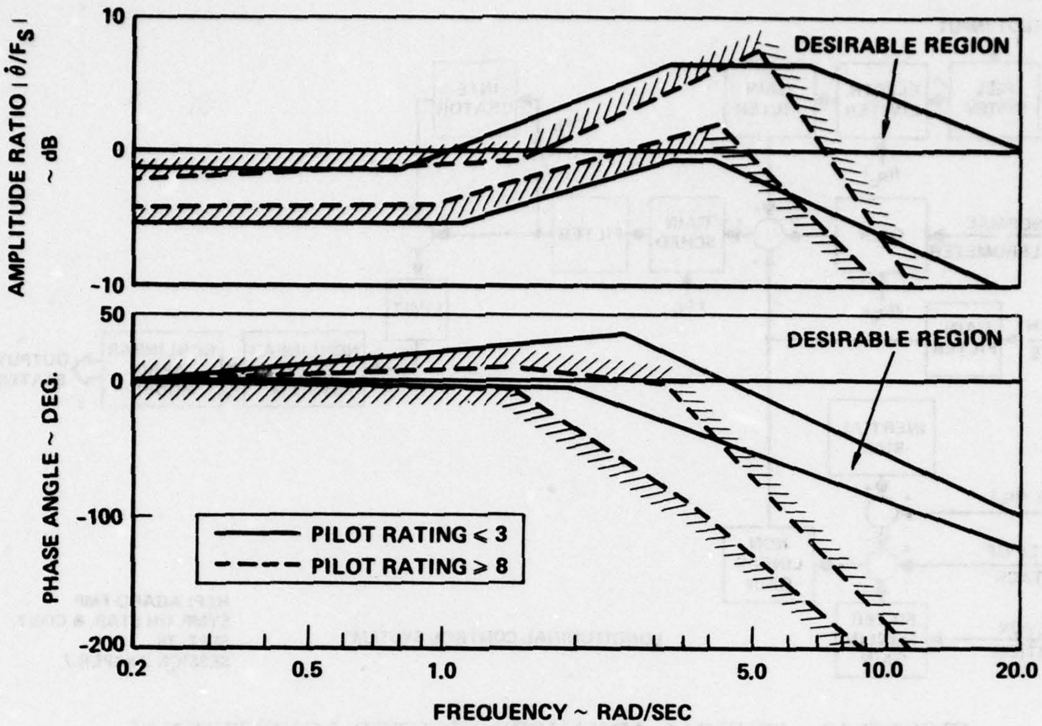

FIGURE 15. DEGRADED FLYING QUALITIES AS THEY APPEAR IN THE FREQUENCY DOMAIN

An equally satisfactory and possibly more direct means of accomplishing flight control system and flying qualities design is to operate directly in the time domain. This approach allows for the effect of plant nonlinearities, allows one to use non-linear elements in the design of the control system itself, and encourages the use of direct digital design of control system software. An example of the non-linearities present in a typical advanced fighter design are shown in Figure 16, taken from reference 7. Figure 17 shows time history responses taken for this system and demonstrates that very well behaved responses can be obtained even for highly unstable airframes. From such time histories classical modal parameters can be estimated using techniques historically used for flight test data reduction. Figure 18 shows the flying qualities of the subject airplane compared to the boundaries of 8785B. In general these results have been varified by piloted flight simulation.

The purpose here is to demonstrate alternatives to the equivalent system approach as stated in reference 1. It is hoped that such methods will be given due consideration in the final selection of a short period response requirement, but regardless of the format of the requirement, it is recommended that the revision

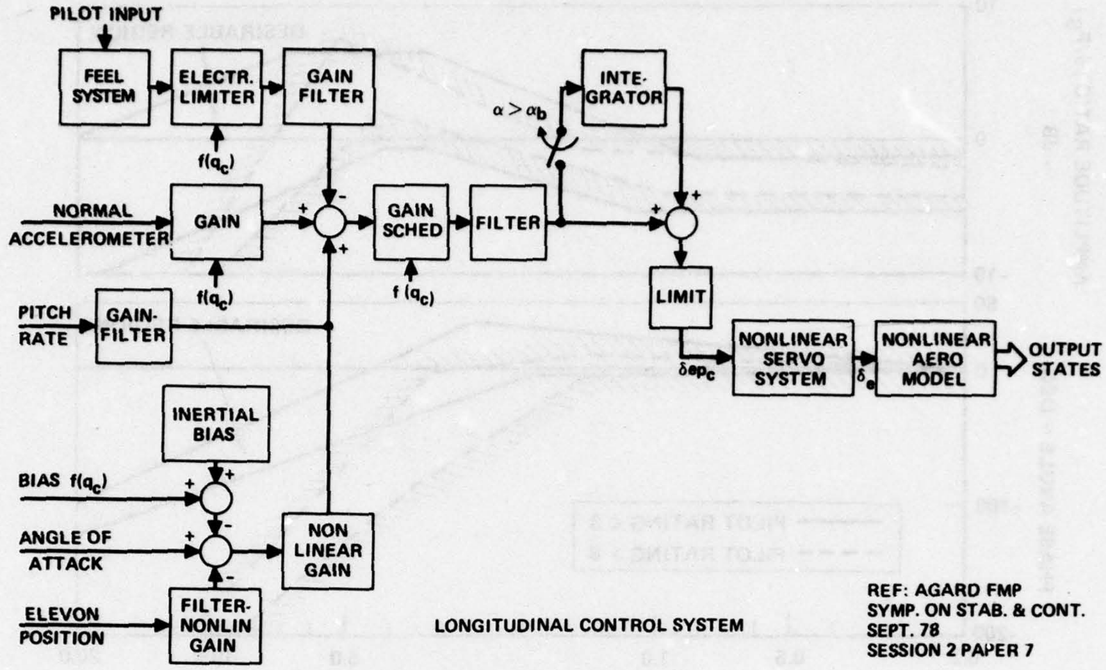

FIGURE 16. TYPICAL ADVANCED FIGHTER LONGITUDINAL FLIGHT CONTROL SYSTEM

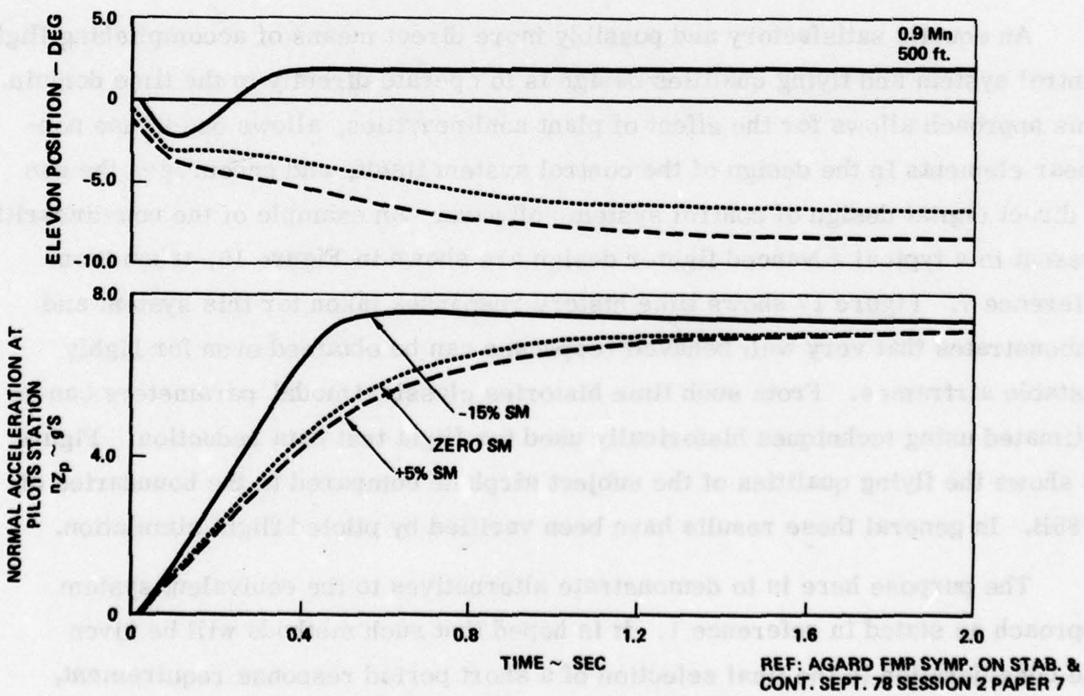

FIGURE 17. TIME RESPONSE EXAMPLE OF HIGHLY AUGMENTED NON-LINEAR AIRCRAFT SYSTEM

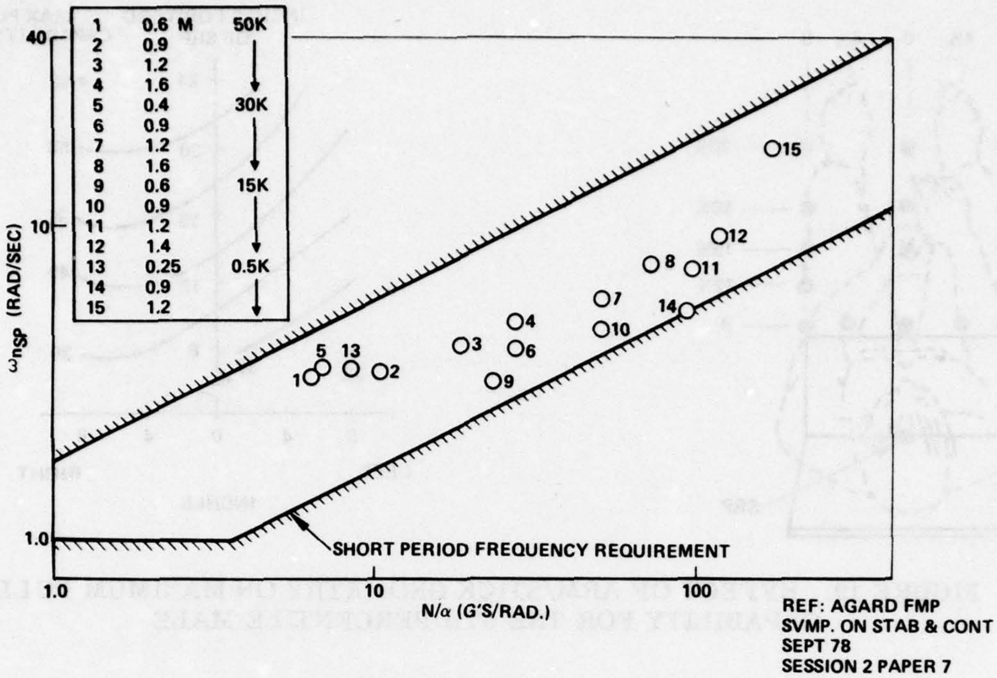

FIGURE 18. HIGHLY AUGMENTED ADVANCED CONFIGURATION COMPARED TO MIL-F-8785B BASED UPON EQUIVALENT CLASSICAL PARAMETERS ESTIMATED FROM SIMULATION TIME HISTORIES

combine the chosen parameters into one criteria. In other words, ζ_{sp} should not be an independent requirement from ω_{sp} vs $n_{z\alpha}$.

CONTROL FEEL IN MANEUVERING FLIGHT (3.2.2.2)

8758B bases it's control requirements on response per pound of stick or pedal force and this is appropriate. However, the specification almost completely ignores the effect of stick position or deflection. Figure 19 shows the effect of arm location relative to the body on the maximum pull capability of a 5th percentile male. These data are from reference 8. They show that one's maximum force capability is not symmetric left and right and varies by about a factor of two for forward and aft stick positions. Figure 20 shows similar data for both pull and push strength as a function of upper arm angle. Here one can see that pull and push strength differ significantly and also that the 5th and 95th percentile male strengths differ by as much as a factor of three.

These data are included here because it is the feeling of the author that their effect is not widely known. It is hypothesized that at least the upper limit of stick

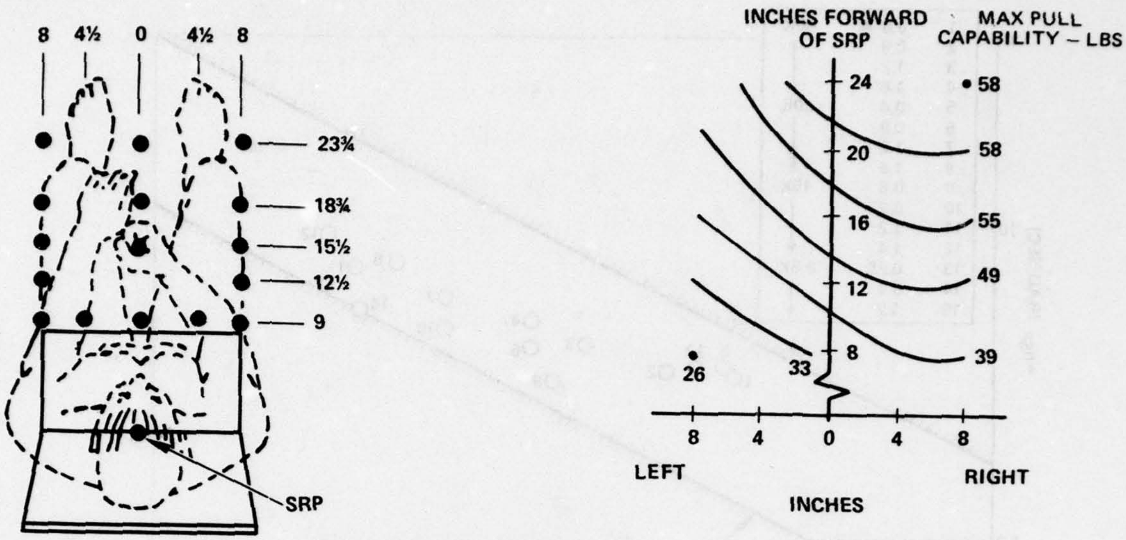

FIGURE 19. EFFECT OF ARM/STICK GEOMETRY ON MAXIMUM PULL CAPABILITY FOR THE 5TH PERCENTILE MALE

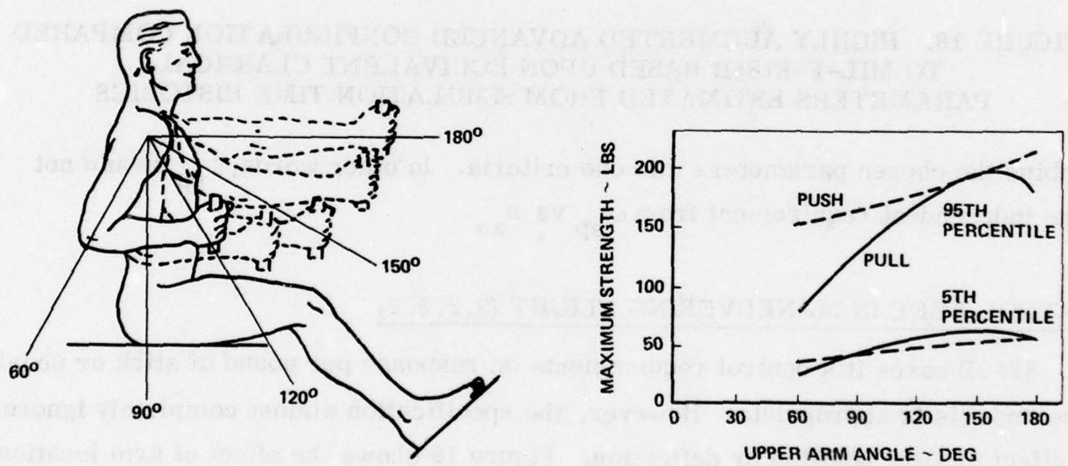

FIGURE 20. EFFECT OF UPPER ARM ANGLE ON PULL AND PUSH STRENGTH FOR THE 5TH AND 95TH PERCENTILE MALE

force should be a function of the percent of pilot effort required and not just of force required at the stick grip. Certainly a given stick force at the grip will feel heavier to the pilot for aft stick positions. Also one must be very careful in correlating the acceptability of stick forces for various aircraft to include the effect of stick location and maximum stick deflection. For instance, the F-5A stick deflection is greater than that of the A-7D by more than a factor of 2. This places the stick in a different location in the cockpit for maximum deflection.

PILOT INDUCED OSCILLATIONS (3.2.2.4)

Northrop's attention was sharply focused on the subject of Pilot Induced Oscillations (PIO) as the result of a very dramatic encounter on the T-38A in 1960. A time history of that incident is shown as Figure 21. Peak load factors ranged between -9 and +8 g's but the aircraft was brought under control and recovered. The incident occurred on a low altitude high-speed run on which the stability augmenter malfunctioned. A limit cycle occurred (not shown in Figure 21) in the pitch SAS and the pilot disengaged the SAS at a peak value of surface command and the aircraft experienced a step input to the stabilizer. This initiated large pilot inputs which coupled with the airplane to form the PIO. See references 9 and 10 for details.

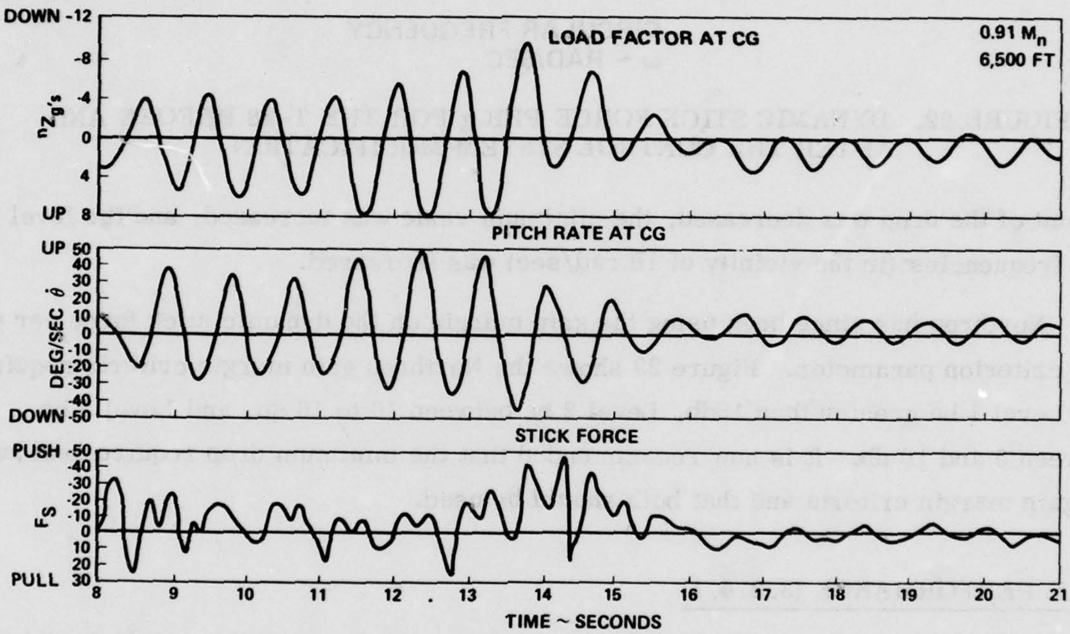

FIGURE 21. A FLIGHT TIME HISTORY OF A T-38A PILOT INDUCED OSCILLATION (JAN. 1960)

The incident was attributed to several factors including the bob-weight, feel-spring, and horizontal tail servo-value. Changes were made to these parameters and the modified versions of the T-38 have been PIO free. The investigation, analysis, and research conducted in support of the T-38 emphasized the importance of the variation in stick force per g versus frequency. This dynamic stick force per g for the T-38 before and after the control system modification is shown in Figure 22. The difficulty is that at least three potentially significant features have changed simultaneously. The

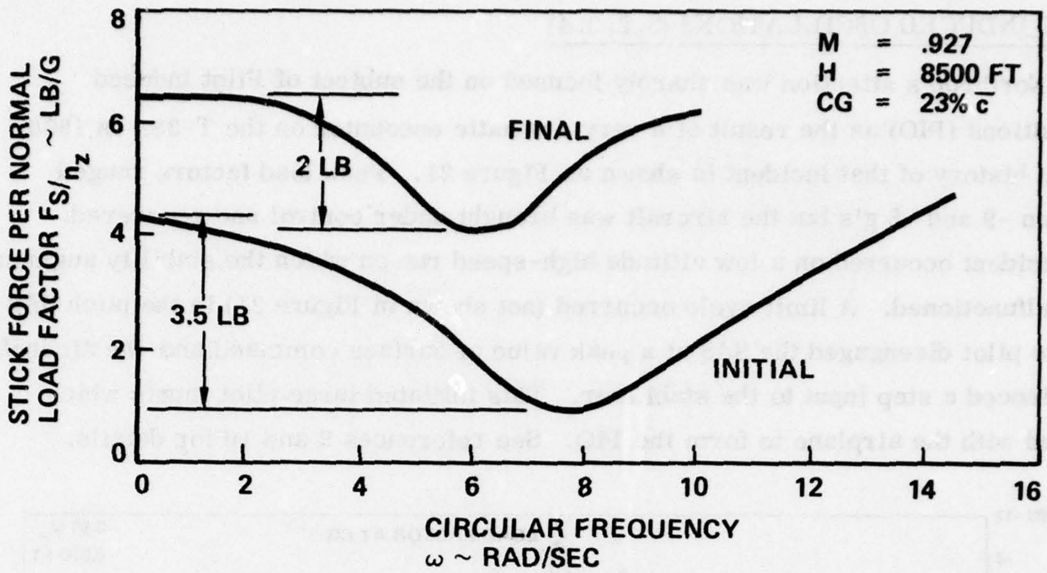

FIGURE 22. DYNAMIC STICK FORCE PER g FOR THE T-38 BEFORE AND AFTER THE CONTROL SYSTEM MODIFICATION

amount of the drop was decreased, the minimum value was increased, and the level at high frequencies (in the vicinity of 10 rad/sec) was increased.

Northrop has since been using the gain margin on the dynamic stick force per g as a criterion parameter. Figure 23 shows the Northrop gain margin criteria requires that Level 1 be greater than 16db, Level 2 be between 10 to 16 db, and Level 3 be between 5 and 10 db. It is now recommended that the minimum drop requirement augment the gain margin criteria and that both should be used.

ROLL PERFORMANCE (3.3.4.1)

The proposed revision to the roll response requirements are intended to account for the inherent reduction in roll response at low and high airspeeds and at elevated load factors while requiring a higher rate of response in the middle of the envelope for one g flight. For the Combat and Ground Attack Flight Phases the requirements are stated in terms of bank-to-bank rolls for any load factor up to $0.8n_L$. The requirement is to change bank angle by 30, 50, 90, or 180 degrees in a time less than or equal to a specified amount.

There are several factors which should be clarified in the context of this requirement. First, "bank-to-bank" generally refers to symmetric maneuvers; for instance, 60 degrees bank-to-bank is from 30 degrees left wing down to 30 degrees

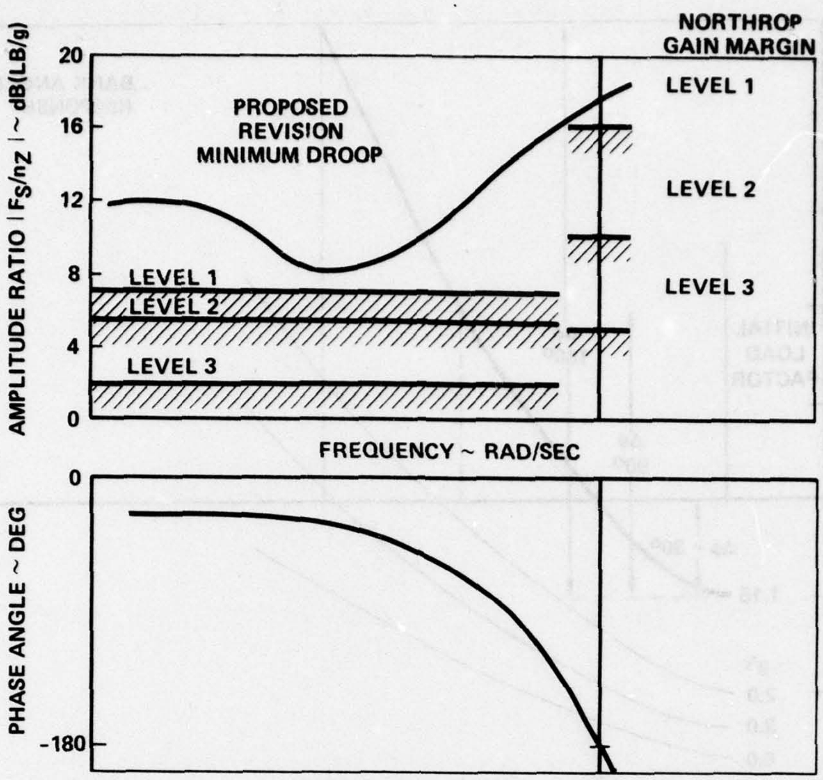

FIGURE 23. DYNAMIC STICK FORCE PER g CRITERIA

right wing down. It is assumed for the purposes of this paper that for the elevated load factor conditions the required change is from the trimmed bank angle for that load factor through whatever change in bank angle is required. See Figure 24. For example, when trimmed at 1.15 g's the change in bank angle of 90 degrees would be from approximately 30 degrees left wing down to 60 degrees right wing down. In addition, especially when considering steep clumbing or diving trajectories or high angle-of-attack conditions, one must be concerned about the definition of bank angle and the axis system in which it is defined. A standard practice is to use the integral of the body axis roll rate. The specification requires that the roll angle be measured in the Y-Z plane between the y-axis and the horizon but is not clear if this is a body or stability axis coordinate system.

The required roll performance is stated for four separate speed ranges from very low (VL) to High (H). These speed ranges were calculated for the F-5E in the Combat (CO) and Ground Attack (GA) Flight Phases. For the speed ranges as defined in reference 1 certain of the speed ranges collapsed to a single value. As shown in

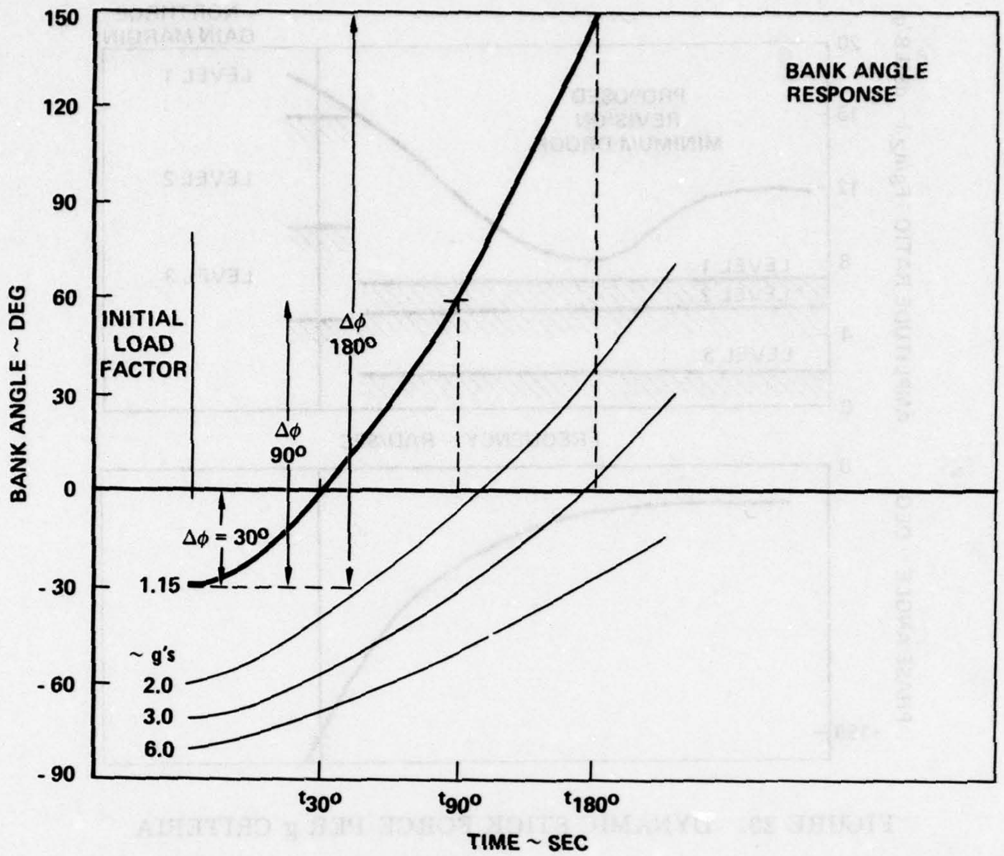

FIGURE 24. ROLL PERFORMANCE FLIGHT RESPONSE MEASUREMENTS

Figure 25 for the CO Flight Phase the VL range collapse and for the GA Flight Phase both the VL and M (Medium) ranges collapsed.

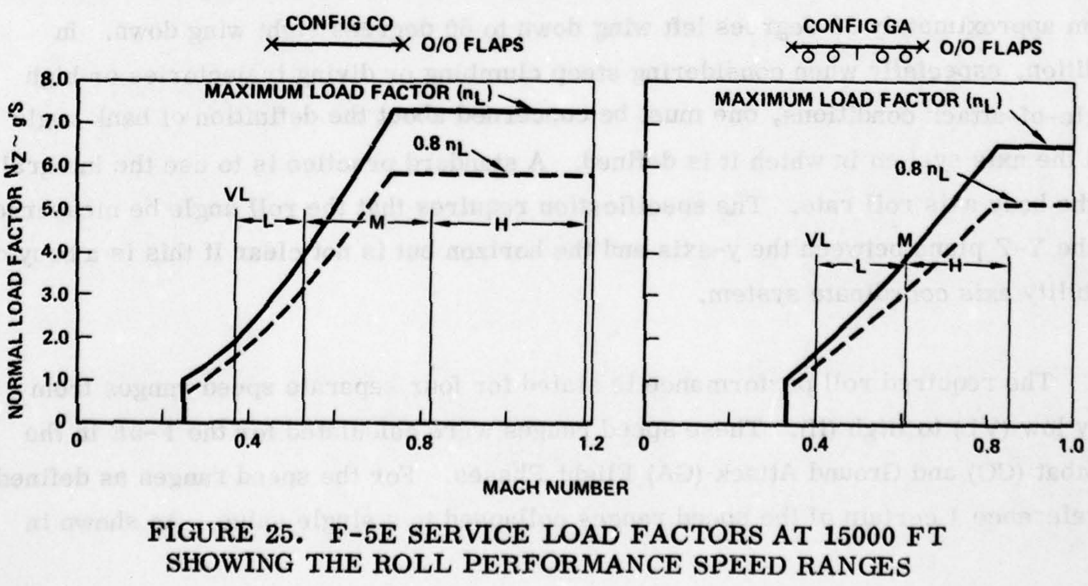

FIGURE 25. F-5E SERVICE LOAD FACTORS AT 15000 FT SHOWING THE ROLL PERFORMANCE SPEED RANGES

It is recommended that the speed ranges be redefined in such a way that this collapsing of ranges does not occur. Figure 26 shows the speed range definitions as stated in the proposed revisions (reference 1) and the Northrop recommended definitions. It is recommended the speed ranges be defined as a percentage of the entire operational speed range. The lower 10% is VL, the next 30% is L, the next 30% is M, and the remaining 30% is H. With these definitions all aircraft are treated equally with the ranges being an "elastic" fit to the particular total speed range.

8785B PROPOSED REVISION

SPEED RANGE SYMBOL	EQUIVALENT AIRSPEED RANGE
VL	$V_o \, MIN \leq V < V_{MIN} + 20 \, KTS$
L	$V_{MIN} + 20 \, KTS \leq V < 2VS$
M	$2V_S \leq V < 0.7 \, V_{MAX}$
H	$0.7 \, V_{MAX} \leq V \leq V_o \, MAX$

NORTHROP RECOMMENDATION

SPEED RANGE SYMBOL	EQUIVALENT AIRSPEED RANGE
VL	$V_o \, MIN \leq V < V_o \, MIN + 0.1 \, \Delta V$
L	$V_o \, MIN + 0.1 \, \Delta V \leq V < V_o \, MIN + 0.4 \, \Delta V$
M	$V_o \, MIN + 0.4 \, \Delta V \leq V < V_o \, MIN + 0.7 \, \Delta V$
H	$V_o \, MIN + 0.7 \, \Delta V \leq V \leq V_o \, MAX$
WHERE $\Delta V = V_o \, MAX - V_o \, MIN$	

FIGURE 26. RECOMMENDED SPEED RANGE DEFINITIONS FOR THE ROLL PERFORMANCE REQUIREMENTS

An example of the F-5E roll performance is shown in Figure 27 which plots time to bank versus Mach number for maximum rolls at $0.8 \, n_L$. The data includes time to roll 30 degrees, 50 degrees, and 90 degrees. The solid symbols show the required maximum time allowable for the bank angle change for each speed range. If the speed ranges are defined as in reference 1 and as shown on the left half of Figure 27, the F-5E does not, in general, meet the requirement. If the speed ranges are defined as recommended by Northrop and as shown in the right half of Figure 27, then the requirements are satisfied for most conditions although not universally. Since the F-5E roll performance is found very satisfactory in operational use, it is recommended that the speed ranges be redefined. The performance levels are then approximately satisfied.

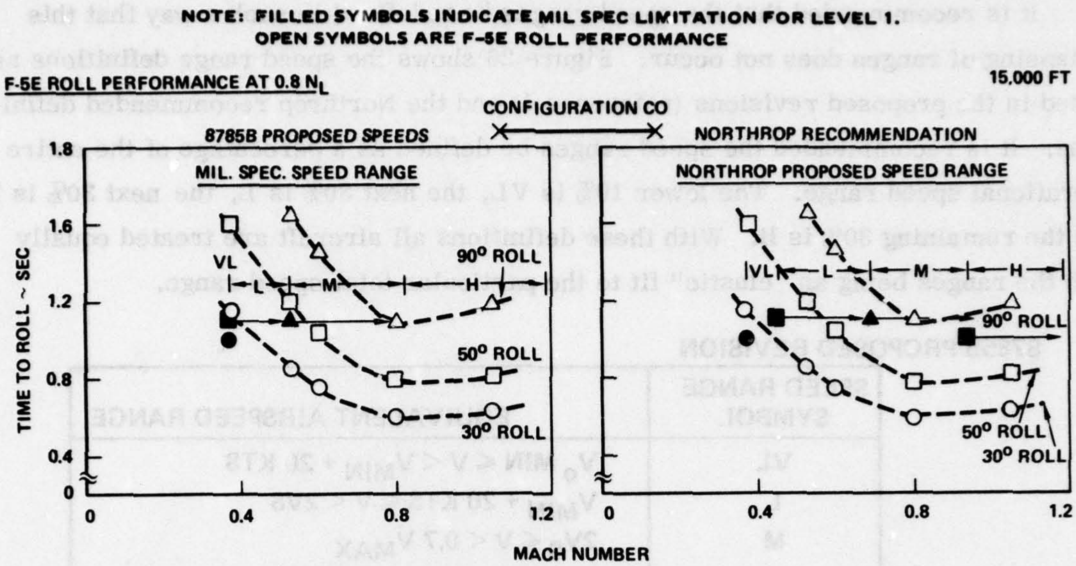

FIGURE 27. F-5E ROLL PERFORMANCE

ATMOSPHERIC DISTURBANCES (3.7)

Northrop has extensive experience in the modeling and use of atmospheric disturbances in both analysis methods and in piloted simulations. See references 4 and 11. The most recent experience has been with respect to Gaussian vs. non-Gaussian turbulence models (ref. 12). The non-Gaussian model employed is known as the Tomlinson (or Jones) Model. The so-called Tomlinson model is featured as being more representative of the true time varying or intermittent properties of the real atmosphere. Figure 28 shows that the Tomlinson model has fewer changes of intermediate velocity with a "greater-than-Gaussian" probability that there will be either very small velocity increments or extremely large velocity changes. Figure 29 shows a comparison of representative Tomlinson and Dryden forms.

Reference 12 presents the results of an analytical and piloted simulation investigation of a ride improvement mode system for the YF-17 in low altitude high speed flight. Both Gaussian and non-Gaussian turbulence models were used. Since reference 12 is available in the open literature one is referred there for details. It was concluded there however that the performance of the flight control system design may be influenced by the choice of turbulence model and that this aspect may need to be reflected in the appropriate military specification.

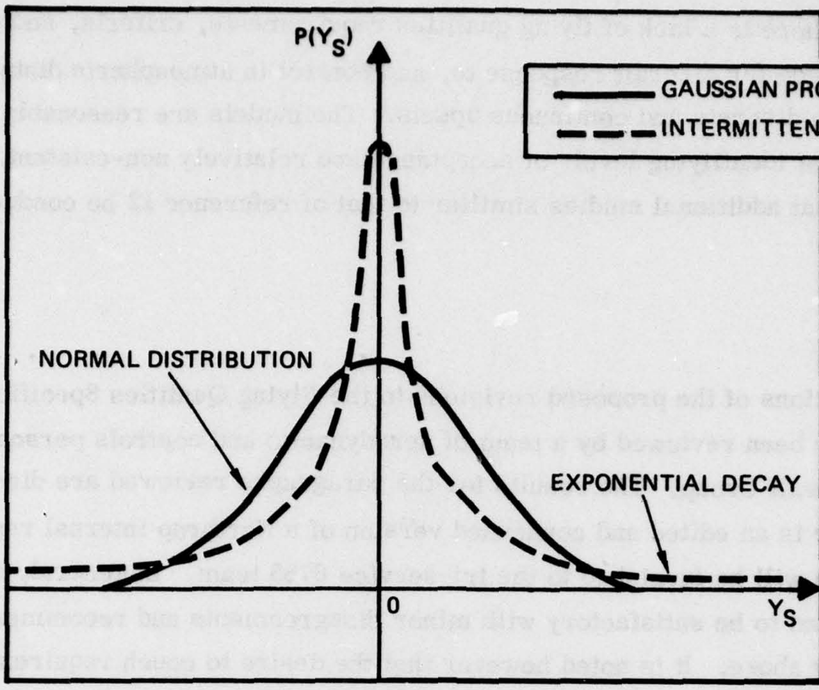

FIGURE 28. PROBABILITY DENSITY OF VELOCITY INCREMENTS (SCHEMATIC)

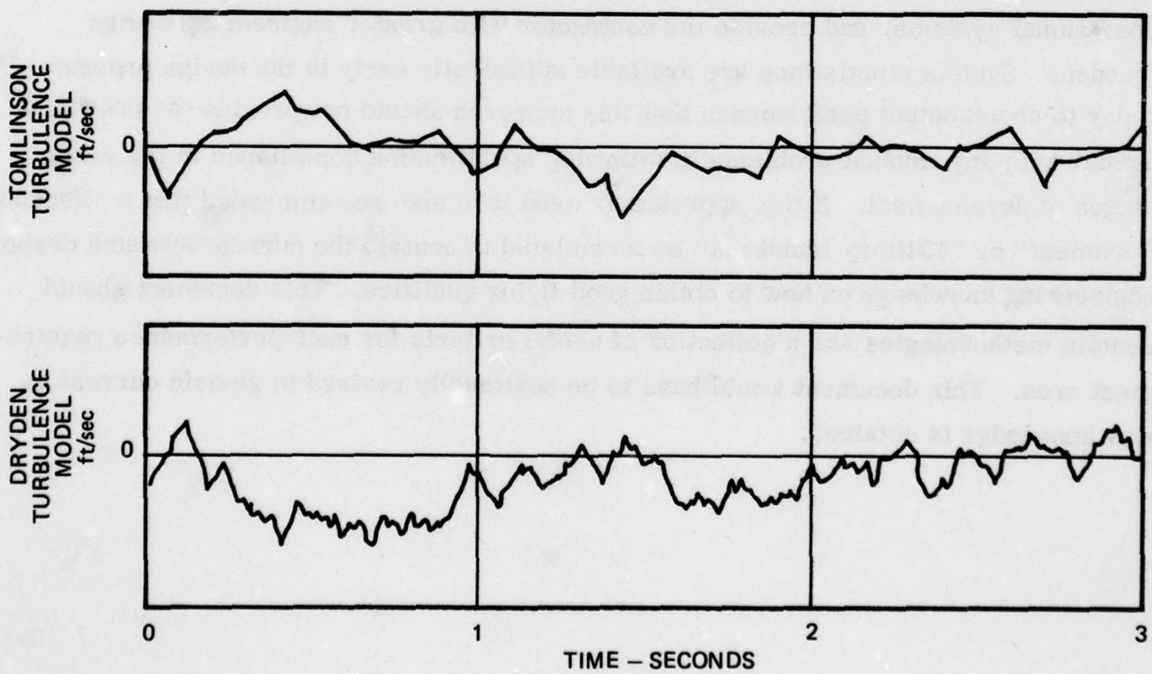

FIGURE 29. W – GUST TIME HISTORIES FOR THE TWO TURBULENCE MODELS

In addition, there is a lack of flying qualities requirements, criteria, and even rough design guidance for aircraft response to, and control in atmospheric disturbances. This is true of both discrete and continuous upsets. The models are reasonably well defined but means of identifying levels of acceptance are relatively non-existent. It is recommended that additional studies similiar to that of reference 12 be conducted to fill this void.

SUMMARY

Selected sections of the proposed revisions to the Flying Qualities Specification, MIL-F-8785B have been reviewed by a team of aerodynamic and controls personnel at Northrop's Aircraft Group. The results for the paragraphs reviewed are discussed above. This paper is an edited and condensed version of a Northrop internal report reference 13 which will be furnished to the tri-service 8785 team. In general, the revisions were found to be satisfactory with minor disagreements and recommendations covered separately above. It is noted however that the desire to couch requirements in terms of engineering parameters of the system itself is leading to an increasingly complex and cumbersome specification. As a philosophical recommendation for future revisions, it is suggested that stating the requirements in terms of the desired output performance should simplify the specification, better insure satisfactory operational systems, and provide the contractor with greater engineering design freedom. System simulations are available sufficiently early in the design process today to check output performance that this approach should not provide the procuring agency with any unusual problems in following specification compliance in the early stages of development. If this approach is used it is also recommended that a "Backup Document" or "Military Handbook" be formulated to contain the current systems design engineering knowledge on how to obtain good flying qualities. This document should contain methodologies and a collection of useful criteria for each performance requirement area. This document would have to be continually revised to remain current as new knowledge is obtained.

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AFFDL 1978 FLYING QUALITIES SYMPOSIUM & WORKSHOP

B-1 EXPERIENCE RELATED TO MIL-F-8785B* AND PROPOSED REVISIONS

BY

J. E. CAMPBELL

The B-1 is a large, flexible, low limit load factor, variable wing sweep aircraft with automatic and manual terrain following and air refueling capabilities.

During the B-1 development program, considerable analytical, flight simulation test, "iron-bird" test and flight test experience has been obtained relative to the flying qualities requirements of MIL-F-8785B and proposed revisions to this specification. Comments concerning experience relative to the requirements of particular paragraphs of MIL-F-8785B are presented below. Suggestions for possible requirements changes or for additional testing and validation are included. The specification paragraphs that are addressed are listed in Chart 1.

3.1.10.2 REQUIREMENTS FOR A/P FAILURE STATES - CHART 2

The requirements on the probability of encountering level 3 H.Q. due to failures are quite stringent. In the B-1, required mission success probability can be achieved with a (FCS) failure probability that is about 10 times that allowed by 3.1.10.2 for encountering level 3 H.Q. Since level 3 H.Q. are safe but probably not adequate for mission success and since, generally speaking, the same failure states that result in level 3 H.Q. also result in mission failure, it is recommended that the level 3 requirements be tied to mission success requirements.

If the initial design of an FCS/airplane combination does not satisfy 3.1.10.2 requirements, it is generally necessary to add more redundancy or some other degree of costly complexity. In order to avoid requirements that result in FCS overdesign, or to permit showing that a given system is not underdesigned, consideration might be given to allowing:

*MIL-F-8785B (ASG) Military Specification, Flying Qualities of Piloted Airplanes.

(1) Use of average instead of maximum mission times in probability calculations. In a multiply redundant system the probability of failure is a function of mission time higher than 1st order. A system designed to meet failure requirements at the maximum mission time will be over designed when considering all missions and mission times.

(2) Including the probabilities of being at flight conditions where level 3 H.Q. are encountered. If certain failure modes result in level 3 H.Q. only at unusual flight conditions or those near the envelope extremes, the FCS will be overdesigned if it is assumed to be always operating at these conditions.

(3) The use of piloted flight simulation to evaluate flying qualities levels associated with various failure states. If certain failure modes degrade flying qualities, based on flight simulation, to a lesser extent than might be predicted from evaluation of specification parameters, then the FCS may be overdesigned unless these cases are recognized.

3.2.2.1.3 RESIDUAL OSCILLATIONS - CHART 3

During early flights on the B-1, pitch axis residual oscillation were observed at certain flight conditions that satisfied 3.2.2.1.3 requirements. At these conditions large amplitude pitch damping also satisfied 3.2.2.1.2 requirements, but pilots commented on inability to make small precise pitch changes. In a system with residual oscillation tendencies, damping ratio is a function of amplitude of control surface or aircraft motion as shown on the sketch of Chart 3. In such a system a pilot might try to make small control inputs in the range that result in poorly damped responses and consequently have difficulty in stabilizing the aircraft. Elimination of the residual oscillation solved the problem in the B-1 and may be the only answer for acceptable flying qualities.

3.2.2.2.1 CONTROL FORCES IN MANEUVERING FLIGHT - CHART 4

Early in the B-1 program various flight simulation programs were run that indicated the MIL-specification lb/g requirements were too high at various mission conditions, especially during terrain following. Specific comments are included on Chart 4. The aircraft FCS was initially set up to provide lb/g lower than specification requirements but subsequent augmentation gain changes, made because of structural mode coupling considerations, have resulted in lb/g characteristics that satisfy $N_L=3$ requirements but are less than $N_L=2$ requirements. During MTF flight testing lb/g values near 17 have been found to be too high based on fatigue considerations. Pilot comments relating to degree of fatigue associated with this force level are given on Chart 4. Future testing with lower force gradients is planned to determine desired levels but it is evident some provision needs to be made for special tasks.

3.3.1.2 ROLL MODE,

3.3.4 ROLL CONTROL EFFECTIVENESS,

3.3.4.2 AILERON CONTROL FORCES - CHART 5

These paragraphs, when used in combination, put some limits on allowed values for steady - state roll rate per roll stick force gradient but meeting these requirements doesn't insure good handling qualities. B-1 roll axis FCS parameter variation flight tests at several mission conditions show that both roll control characteristics considered sensitive and those considered to have high forces fall within allowed specification limits. Roll rate sensitivity, $\dot{\phi}/F$, seems to be a sufficiently important parameter to warrant a direct specification of allowed limits.

3.5.3 DYNAMIC CHARACTERISTICS - CHART 6

The intent of this paragraph is to insure a control system that has adequate dynamic response to allow favorable aircraft response control. It would seem logical to establish requirements on control force to aircraft response dynamics rather than on the FCS only. This is done for the pitch axis in the proposed paragraph 3.2.2.4.3. If similar paragraphs for the roll and yaw axes were established, 3.5.3 could be deleted.

The accompanying sketch on Chart 6 shows the δ/F transfer function for the B-1 that should be evaluated to show compliance with specification requirements rather than the open-loop stick force to surface transfer function that would be evaluated in an unaugmented aircraft. Characteristics that might be unsatisfactory open loop (no aero feedbacks) may be satisfactory when used with the aero feedbacks. In fact, during closed loop operations, stick force to surface phase angles may show lead rather than lag.

3.5.5.1 FAILURE TRANSIENTS - CHART 7

No B-1 in-flight augmentation failure transient experience has been obtained, but piloted flight simulation failure studies, based on aircraft hardware failure transient characteristics, provide some validation for proposed, 0.5g in 2 seconds allowable transient values in the pitch and yaw axes. Roll axis failures were too small to provide any validation. Simulation tests run at air refueling and terrain following test conditions showed that first and second pitch and yaw augmentation failure load factor transients are readily controlled by the pilot. Since level 3 flying qualities exist after second pitch and yaw augmentation failure transients (0.5g in 2 seconds) and since the pilots readily controlled these failure transients, it is reasonable to assume they could also have controlled the same failures with a level 1 or 2 flying qualities aircraft. Thus, for normal operations (not near the ground) the proposed allowable load factor transients seem acceptable. During terrain following, altitude loss may become a problem so that early failure detection is important.

During MTF operations, a pitch axis second augmentation failure is essentially unnoticed since the failure transient reflects on the TF display and is corrected by pilot action. During ATF operations, a second augmentation failure cannot be automatically responded to, since the automatic mode operates through the augmentation, and aircraft safety must be provided by pilot control through the mechanical control system. Failure detection time in the 1.5 to 2 seconds range has provided required safety for this type failure in the B-1.

3.5.6.1 TRANSFER TRANSIENTS - CHART 8

In the B-1, the stability augmentation can be shut off to allow pilot familiarization with unaugmented flight characteristics. Such shutoff results in the same control configuration as would exist following two augmentation failures in any one axis. At some flight conditions, this unaugmented configuration may provide level 3 flying qualities. At these conditions transients would satisfy failure state allowables but might not satisfy proposed transfer transient allowables. Perhaps the allowed transients should be based on the F.Q. level expected after the transfer.

NEW REQUIREMENTS - STRUCTURAL MODE COUPLING - CHART 9

During B-1 terrain following flight simulation studies, structural mode characteristics were included. It was found that pilot pitch control inputs cause structural mode motions at the cockpit which interfere with the pilot's control capability. The pilot is not trying to control the structural mode motion or control at the structural mode frequency but just unintentionally excites the modes. Modal motions are poorly damped and die out slowly. Pilot ratings improved significantly when a control input filter was used that prevented excitation of the structure or effectively decoupled the modal motions from the pilot inputs. Similar pilot/lateral structural mode coupling had been observed in earlier AMSA work. Some sort of new pilot/structural mode coupling requirement may be indicated as a result of these experiences.

AD-A066 493 AIR FORCE FLIGHT DYNAMICS LAB WRIGHT-PATTERSON AFB OHIO F/G 1/2

PROCEEDINGS OF AFFDL FLYING QUALITIES SYMPOSIUM HELD AT WRIGHT --ETC(U)

DEC 78 G T BLACK, D J MOORHOUSE, R J WOODCOCK

UNCLASSIFIED AFFDL-TR-78-171 NL

4 OF 7

AD A066493

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

In a related phenomenon, it was also found that pilot ratings and control improved when structural mode motions were filtered out of TF commands displayed to the pilot.

FLYING QUALITIES REQUIREMENTS SYMPOSIUM
ROCKWELL LOS ANGELES COMMENTS
RELATED B-1 EXPERIENCE

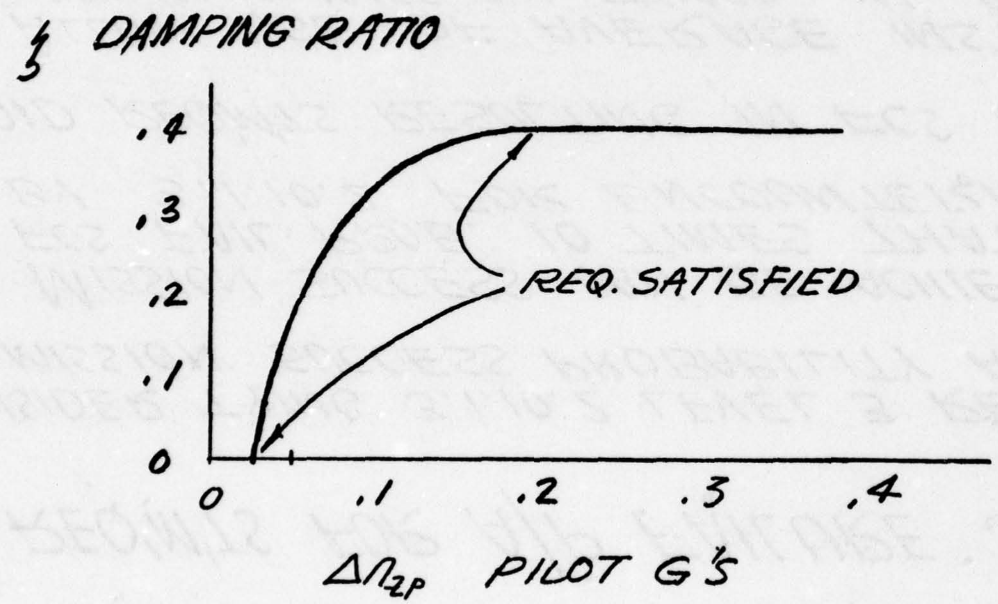
PARA. NO.	TITLE
3.1.10.2	REQM'TS FOR AIRPLANE FAILURE STATES
3.2.2.1.3	RESIDUAL OSCILLATIONS
3.2.2.2.1	CONTROL FORCES IN MANEUVERING FLIGHT
3.3.1.2	ROLL MODE
3.3.4	ROLL CONTROL EFFECTIVENESS
3.3.4.2	AILERON CONTROL FORCES
3.5.3	DYNAMIC CHARACTERISTICS
3.5.5.1	FAILURE TRANSIENTS
3.5.6.1	TRANSFER TRANSIENTS
NEW	STRUCTURAL MODE COUPLING.

3.1.10.2 REQ'M'TS FOR A/P FAILURE STATES

- CONSIDER TYING 3.1.10.2 LEVEL 3 REQ'M'TS TO MISSION SUCCESS PROBABILITY ALLOWABLES
- B-1 MISSION SUCCESS CAN BE ACHIEVED WITH FCS FAIL. PROB. 10 TIMES THAT ALLOWED BY 3.1.10.2 FOR ENCOUNTERING LEVEL 3.
- AVOID REQ'M'TS RESULTING IN FCS OVERDESIGN
  - ALLOW USE OF AVERAGE INSTEAD OF MAXIMUM MISSION TIMES IN PROBABILITY CALCULATIONS
  - ALLOW INCLUSION OF PROBABILITIES OF OPERATING AT FLIGHT CONDITIONS WHERE FAILURE MODES RESULT IN LEVEL 3 H.Q.
  - ALLOW USE OF PILOTED FLIGHT SIMULATION FOR DETERMINING H.Q. LEVELS ASSOCIATED WITH VARIOUS FAILURE STATES

3.2.2.1.3 RESIDUAL OSCILLATIONS

- EARLY B-1 FLIGHTS – RESIDUAL OSCILLATIONS SATISFIED 3.2.2.1.3 REQ'M'TS
- LARGE AMPLITUDE PITCH DAMPING SATISFIED 3.2.2.1.2 REQ'M'TS
- PILOTS COMMENTED ON INABILITY TO MAKE SMALL PRECISE PITCH CHANGES
- REQ'M'T REVISION MAY BE INDICATED



3.2.2.2.1 CONTROL FORCES IN MANEUVERING FLT.

- NO PROVISION MADE FOR SPECIAL TASKS LIKE TERRAIN FOLLOWING
- CONSOLIDATED PILOT COMMENTS - BASED ON FLIGHT SIMULATOR TESTS

  21 LB/G LEVEL 1 MIN. IS TOO HIGH FOR 2G A/C
  7-8 LB/G LEVEL 1 MIN. IS GOOD DESIGN GUIDE FOR 3G A/C
  4.4 LB/G MAY BE CONS. FOR TRANSIENT OP.
  3 LB/G MIN. SHOULD BE MAINTAINED FOR FAILURE MODES

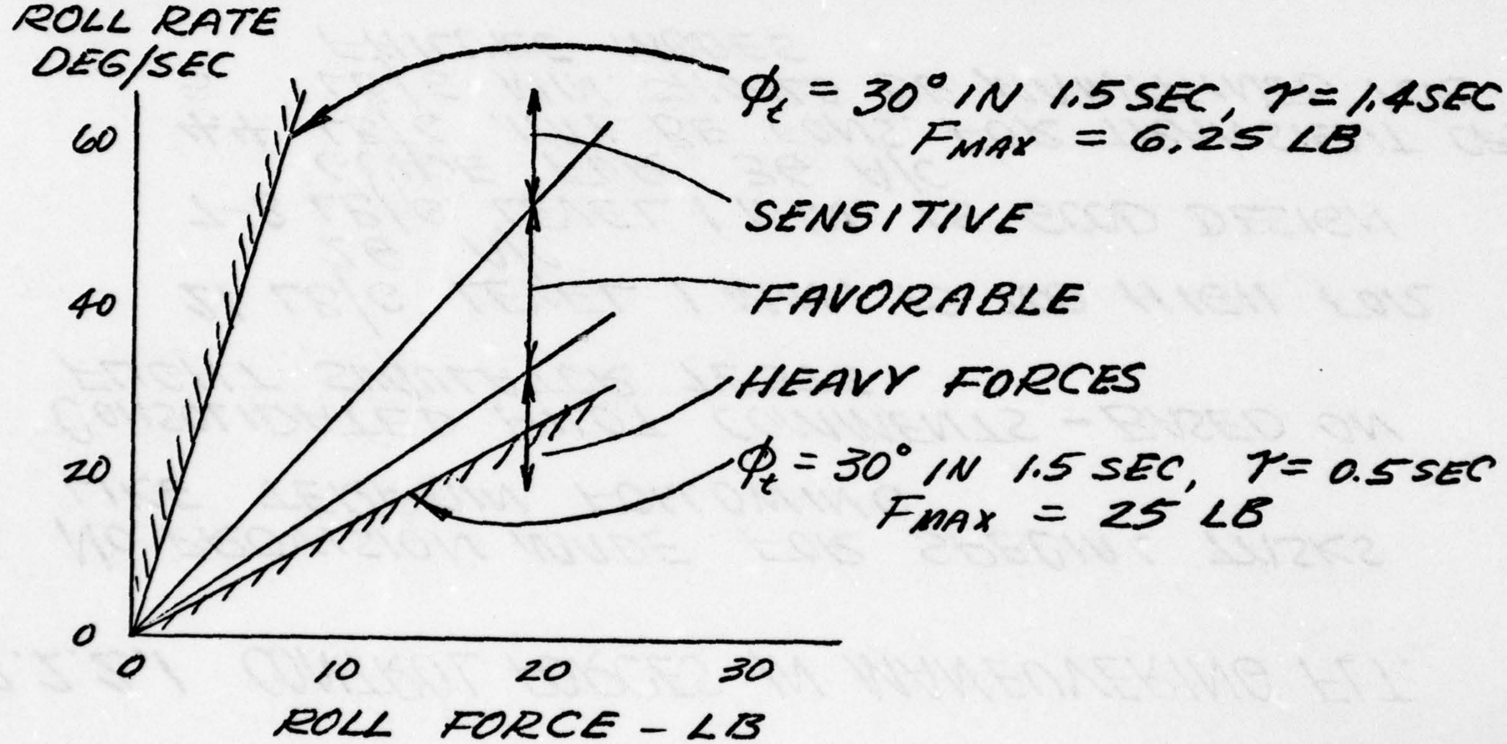
- B-1 MTF FLIGHT PILOT COMMENTS

  17 LB/G TOO HIGH BASED ON FATIGUE
  MEETS $n_L = 3$ REQMTS
  BELOW $n_L = 2$ REQMTS
  OVER RUGGED TERRAIN - 10 MIN. IS TIRING
  COMPOSITE TERRAIN - 2 PILOTS SHARING TASK
  SHORT TASK - 30 MIN.
  MEDIUM TASK - 1 HR
  LONG TASK - 2 HR

- REDUCED FORCE GRADIENTS TO BE TESTED

3.3.1.2 ROLL MODE
3.3.4 ROLL CONTROL EFFECTIVENESS
3.3.4.2 AILERON CONTROL FORCES

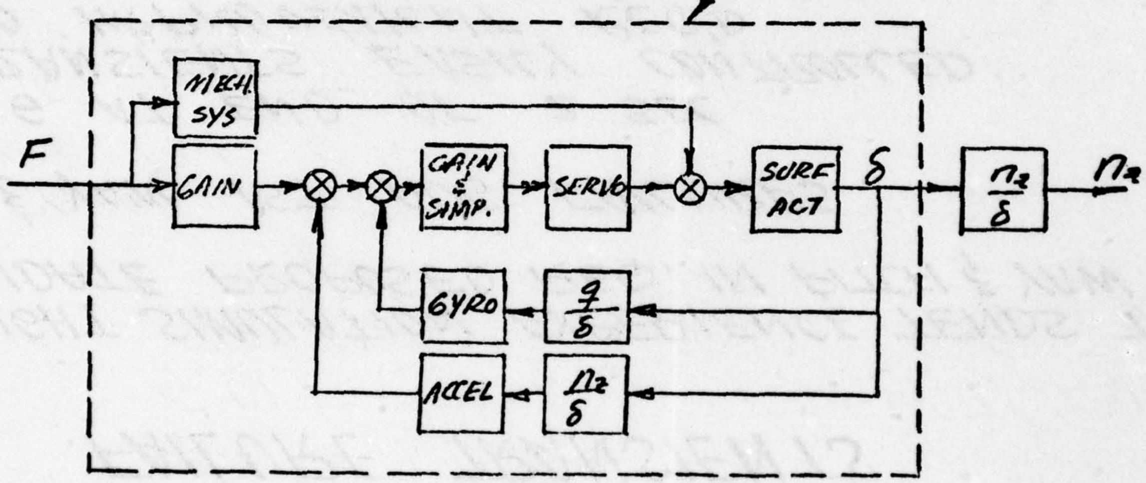
- THESE PARAGRAPHS TEND TO DEFINE ALLOWED RANGE FOR $\dot{\phi}/F$
- MEETING REQMTS DOESN'T INSURE GOOD H.Q. IMPORTANCE OF $\dot{\phi}/F$ MAY JUSTIFY DIRECT SPECIFICATION OF ALLOWABLE VALUES



3.5.3 DYNAMIC CHARACTERISTICS

- PITCH AXIS PHASE REQ. COVERED BY PROPOSED PARA. 3.2.2.4.3
- REQ'MTS FOR ROLL & YAW AXES CORRES. TO THOSE IN PROPOSED PARA. 3.2.2.4.3 WOULD BE MORE MEANINGFUL THAN THOSE IN 3.5.3
- WITH THESE CHANGES, 3.5.3 COULD BE DELETED
- AS 3.5.3 STANDS AUG. A/C CLOSED LOOP $\angle\frac{\delta}{F}$ PHASE ANGLES MUST BE USED TO SHOW COMPLIANCE — NOT OPEN LOOP $\angle\frac{\delta}{F}$

AUGMENTED B-1 $\frac{\delta}{F}$



3.5.5.1 FAILURE TRANSIENTS

- B-1 FLIGHT SIMULATION EXPERIENCE TENDS TO VALIDATE PROPOSED REQ. IN PITCH & YAW AXES
- PITCH & YAW 1ST AUG. FAILURES

  .2G AT END OF 2 SEC
  TRANSIENTS EASILY CONTROLLED
  NO IMPROVEMENT REQ'D

- PITCH & YAW 2ND AUG. FAILURES

  .5G AT END OF 2 SEC.
  TRANSIENTS EASILY CONTROLLED
  NO IMPROVEMENT REQ'D DURING MTF.
  FAILURE DETECTION TIME IMPROVEMENT REQ'D DURING ATF BECAUSE OF POSSIBLE ALTITUDE LOSS

- ROLL AUG 2ND FAILURES

  2 DEG/SEC AT END OF 2 SEC
  TOO SMALL TO HELP VALIDATE PROPOSED REQMT.

3.5.6.1 TRANSFER TRANSIENTS

- REQM'T DOESN'T ALLOW FOR SPECIAL CASES
- B-1 AUGMENTATION CAN BE SHUT OFF FOR TRAINING PURPOSES
- AUG. SHUTOFF CORRESPONDS TO DOUBLE FAILURE
- CRITICAL FLT. COND. TRANSIENTS SATISFY FAILURE REQM'TS BUT EXCEED PROPOSED 3.5.6.1 ALLOWABLES
- CONSIDER RELATING ALLOWED TRANSIENTS TO H.Q. LEVEL AFTER TRANSFER

NEW REQ'M'T - STRUCTURAL MODE COUPLING

- B-1 ABRUPT MTF PITCH CONTROL INPUTS CAUSE STRUCTURAL MODE EXCITATION AT COCKPIT
- SIMULATION TESTS SHOW DEGRATION OF PILOT CONTROL CAPABILITY & RATINGS DUE TO PILOT/STRUCTURAL MODE COUPLING
- RATINGS SIGNIFICANTLY IMPROVED BY DECOUPLING USING PILOT CONTROL INPUT FILTER.
- SIMILAR PILOT/LATERAL STRUCTURAL MODE COUPLING OBSERVED IN EARLY AMSA WORK
- NEW PILOT/STRUCTURAL MODE COUPLING REQ'M'T MAY BE INDICATED

Chick Chalk, Calspan: Does the aircraft have a 'g' limiter?
Answer: Not in the manual terrain following mode. No g limiter in any manual mode.

Tim Sweeney, ASD: Is it the forces or task of terrain following that is tiring?
Answer: Pilots complain of tired wrists.

Question: What is the structural mode frequency that was the problem?
Answer: The lowest fuselage vertical bending mode occurs at about $3H_z$.

Question: What is the frequency of residual oscillation?
Answer: Approximately 0.5 H_z. Aircraft pitch short period frequency is in the range of 2-4 rad/sec.

Question: What is the nature of the pilot input filter?
Answer: Pilot electrical control input filters in the pitch & roll axes use first order lags with a time constant of 0.5 sec. A notch filter was used on the input to the terrain following display.

AN APPROACH TO SIMPLIFY THE SPECIFICATION OF LOW SPEED MANEUVERING PITCH CONTROL FORCE

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It is shown that for an irreversible pitch control system at low speeds, the use of the pitch control force gradient parameter F_s/N is inappropriate. The pitch control force gradient variation with speed at low Mach numbers is discussed along with the maximum load factor variation with speed. Typical maneuvering control force characteristics with load factor are given for several aircraft to illustrate the point. Recommendations are put forth for the specification of low speed maneuvering pitch control forces.

The emphasis of maneuvering characteristics has generally been in the mid or high speed range for most aircraft. In many cases the low speed or high angle of attack range is important also. However, the criterion for maneuvering characteristics developed for the higher speeds cannot be applied to the low speed range due to some fundamental differences.

The variation of the maneuvering force gradients is developed below for an irreversible control system. From the linearized longitudinal lift and moment equations, assuming constant coefficient data, the elevator control deflection for steady maneuvering is given by

$$\delta_e = \frac{C_{M_\alpha}\left[-(C_L)_{\alpha=0} + \frac{2W}{\rho V_T^2}\frac{N_z}{S_W}\right] + C_{L_\alpha}\left[(C_M)_{\alpha=0} + C_{M_q}\frac{qc}{2V_T}\right]}{C_{M_\alpha} C_{L_{\delta_e}} - C_{L_\alpha} C_{M_{\delta_e}}} \tag{1}$$

and

$$\frac{\partial\delta_e}{\partial\delta N} = \frac{C_{M_\alpha}\frac{2W}{\rho V_T^2 S_W} + C_{L_\alpha} C_{M_q}\frac{\bar{c}}{2V_T}\frac{\partial q}{\partial N}}{C_{M_\alpha} C_{L_{\delta_e}} - C_{L_\alpha} C_{M_{\delta_e}}} \tag{2}$$

Assuming turning flight is more representative of low speed maneuvering, pitch rate is given by,

$$q = \frac{g}{V_T}\left(N - \frac{1}{N}\right) \tag{3}$$

$$\frac{\partial q}{\partial N} = \frac{g}{V_T}\left(1 + \frac{1}{N^2}\right) \tag{4}$$

Then,

$$\frac{\partial\delta_e}{\partial N} = \frac{C_{M_\alpha}\frac{2W}{\rho V_T^2 S_W} + C_{L_\alpha} C_{M_q}\frac{\bar{c}g}{V_T^2}\left(1 + \frac{1}{N^2}\right)}{C_{M_\alpha} C_{L_{\delta_e}} - C_{L_\alpha} C_{M_{\delta_e}}} = f\left(\frac{1}{V_T^2}\right) \tag{5}$$

The control force gradient is given by

$$\frac{\partial F_s}{\partial\delta N} = \left(\frac{dF_s}{d\delta_e}\right)_{\substack{\text{feel}\\ \text{spring}}} \cdot F\left(\frac{1}{V_T^2}\right) \tag{6}$$

which indicates that control force gradients increase as speed is decreased as shown in Figure 1. For a pure manual control system with linear aerodynamic characteristics, it can be shown that

$$\frac{\partial F_s}{\partial N} = \frac{1}{\partial} \rho V_T^2 S_e C_e G_e \left[C_{n\delta_e} \frac{\partial \delta_e}{\partial N} + C_{n\alpha_H} \frac{\partial \alpha_H}{\partial N} \right] \tag{7}$$

where

$$\frac{\partial \alpha_H}{\partial N} = \frac{\partial \alpha}{\partial N} \left(1 - \frac{\partial \epsilon}{\partial \alpha}\right) + \frac{g \ell_t}{V_T^2} \left(1 + \frac{1}{N^2}\right) \tag{8}$$

and

$$\frac{\partial \alpha}{\partial N} = \frac{-C_{L\delta_e} \frac{\partial \delta_e}{\partial N} + \frac{2\,W}{\rho V_T^2\, S_w}}{C_{L\alpha}} = f\left(\frac{1}{V_T^2}\right) \tag{9}$$

The partial derivative $\partial\delta_e/\partial N$ is the same expression developed above (equation 5). Again, in terms of functions of true airspeed, the force gradient is

$$\frac{\delta F_s}{\partial N} = f(V_T^2) \cdot f\left(\frac{1}{V_T^2}\right) = \text{constant with } V_T \tag{10}$$

For a spring tab system the expression for the maneuvering gradient contains an additional term which contains the spring constant as follows

$$\frac{\partial F_s}{\partial N} = f(V_T^2) \cdot \left[f_1\left(\frac{1}{V_T^2}\right) + f_2\left(\frac{1}{V_T^2 + K_s}\right) \right] \tag{10a}$$

Depending on the spring constant value, a non-linear variation of the force gradient occurs with speed. The gradient variation is small though for typical spring constants employed. In the practical case, non-linearities of aerodynamic characteristics with angle of attack and surface deflections usually further increase the force gradients as speed is decreased.

Except for aircraft with minimum speeds limited by control surface deflection or heavy buffet, any additional force at stall speed produces no further g increase and, thus, the maneuvering gradient is infinite. The g capability, however, is decreasing with decrease in speed as shown in Figure 2.

The maximum g available at low speeds is given by

$$N_{MAX} = \left(\frac{C_{L_{MAX}} S\, 1/2\rho}{W}\right) V_T^2 \qquad (11)$$

Of course, at constant altitude $C_{L_{MAX}}$ may vary slightly with Mach number at low speeds but the major variation in g is controlled by V_T^2. For the ideal case (linear characteristics) the control force is constant at maximum lift since

$$F_s = \frac{F_s}{N}(N_{MAX} - 1) = f\left(\frac{1}{V_T^2}\right) \cdot f(V_T^2) = \text{constant} \qquad (12)$$

Again, in many cases, non-linear aerodynamic characteristics produce non-linear forces with g, generally increasing non-linear force. In some airplanes this is desirable as an indicator of stall or stall buffet.

Use of the parameter force gradient, F_s/N, becomes less useful in the speed range where an aircraft is lift limited. The intent of the parameter, F_s/N for the maximum gradient is to insure that the maximum control force at N_L, the design limit load factor, is within the pilot's strength capability, as given by

$$F_{s_{MAX}} = \frac{F}{N}(N_L - 1) \qquad (13)$$

Earlier specifications as well as the current version of MIL-F-8785B (at high values of N/α) use for the maximum allowable stick force gradient

$$\frac{F_s}{N} = \frac{56}{N_L - 1} \quad \text{(center stick controllers)} \qquad (14)$$

where the value of 56, in equation (14), represents a maximum one arm pilot effort in pounds. During the period of time, the OV-10A aircraft was in the flight test stage, it was proposed in reference (a) that at low speeds the specification of the maximum force gradient in MIL-F-8785 be revised. The applicable requirement and the reference (a) proposed limits are shown in Figure 3.

The current MIL-F-8785B specification for the level 1 maximum gradient, requires not more than 28 pounds per g or $240/(N/\alpha)$ lbs/g for center stick controllers and 120 lbs/g or $500/(N/\alpha)$ for wheel controllers. Reference

(b) incorporated the basic idea of allowing force gradient increase at low speeds or low values of N/α, but without supporting data set a maximum value of 28 lbs/g for center stick controllers (level 1) and 120 lbs/g for wheel controllers.

Figure 4 illustrates the control force maneuvering gradient for the OV-10A airplane, which employs a reversible manual control system. A misleading and incomplete picture of the low speed maneuvering forces is obtained from the force gradient variation in Figure 4. However, as shown in Figure 4, as speed is reduced less maximum load factor, N, is available as the force gradients are increasing. In actuality, the peak control force at the accelerated stall is reduced also with speed reduction as shown in Figure 5.

Similar results occur for irreversible control systems as shown in Figure 6 for the XFV-12A aircraft which are based on estimated data. Again, the force gradient shows a misleading picture while the maximum force at the lift limited speed shows nearly a constant force level.

Since combat maneuvering in current and future fighter aircraft is expected to occur also at high angle of attack near stall in combat and tactical maneuvers, this flight regime should be adequately covered in the specification. The control sensitivity is currently handled by the minimum force gradient which is applicable at all speeds.

Therefore, it is recommended that at low values of N/α, the current requirements for maximum force gradient defined as $X/(N/\alpha)$ for level 1 and 2 continue to apply at lower values of N/α as shown in Figure 7. The level 3 maximum force gradient should terminate where it intersects the level 2 boundary. In addition, at all conditions where the operational flight envelope is set by other than the limit load factor, the maximum allowable pitch control force should be as shown below for the load factors specified in para. 3.2.3.2.

Center Stick Controllers

Level	Max Force at $N_O(+)$
1	56
2	85
3	85

Wheel Controllers

Level	Max Force at $N_O(+)$
1	120
2	182
3	182

If sustained maneuvering is required in turning flight, the procuring agency should specify a different value for the maximum stick forces than those recommended above or allow the use of trim to reduce pitch

control forces during the sustained maneuver. At the limits of the permissible flight envelope, pitch control forces should be allowed to increase to any pull force.

<u>References</u>

(a) Chalk, C. R., et al, Recommendations for Revision of MIL-F-8785(ASG), "Military Specification - Flying Qualities of Piloted Airplanes," draft dated 20 March, Revised 26 May 1967, Cornell Aeronautical Laboratory, Inc.

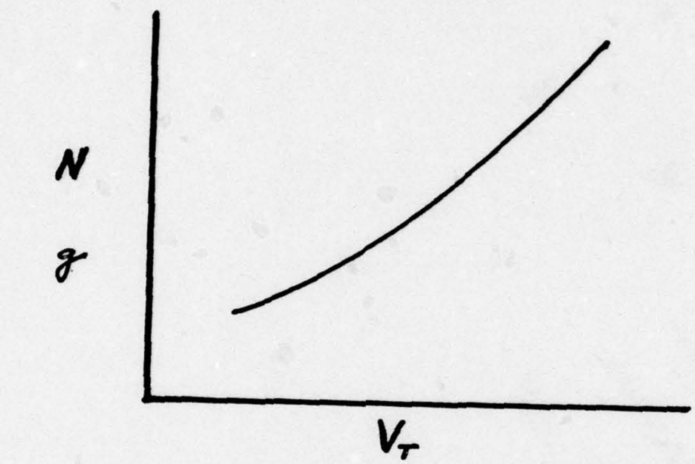

FIG. 2

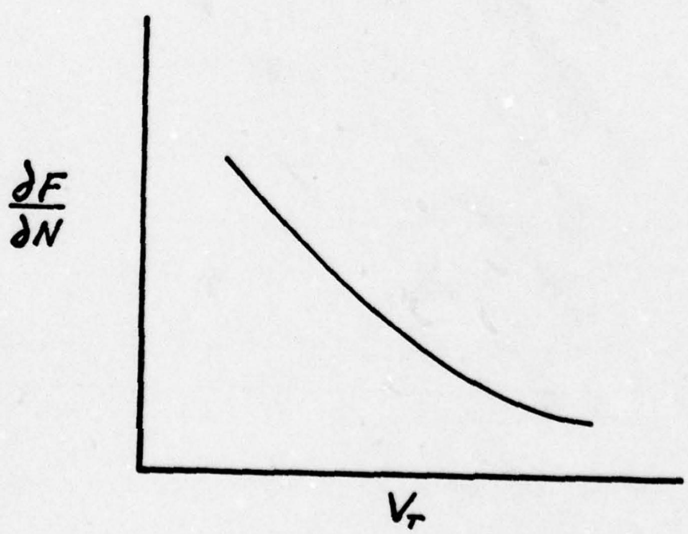

FIG. 1

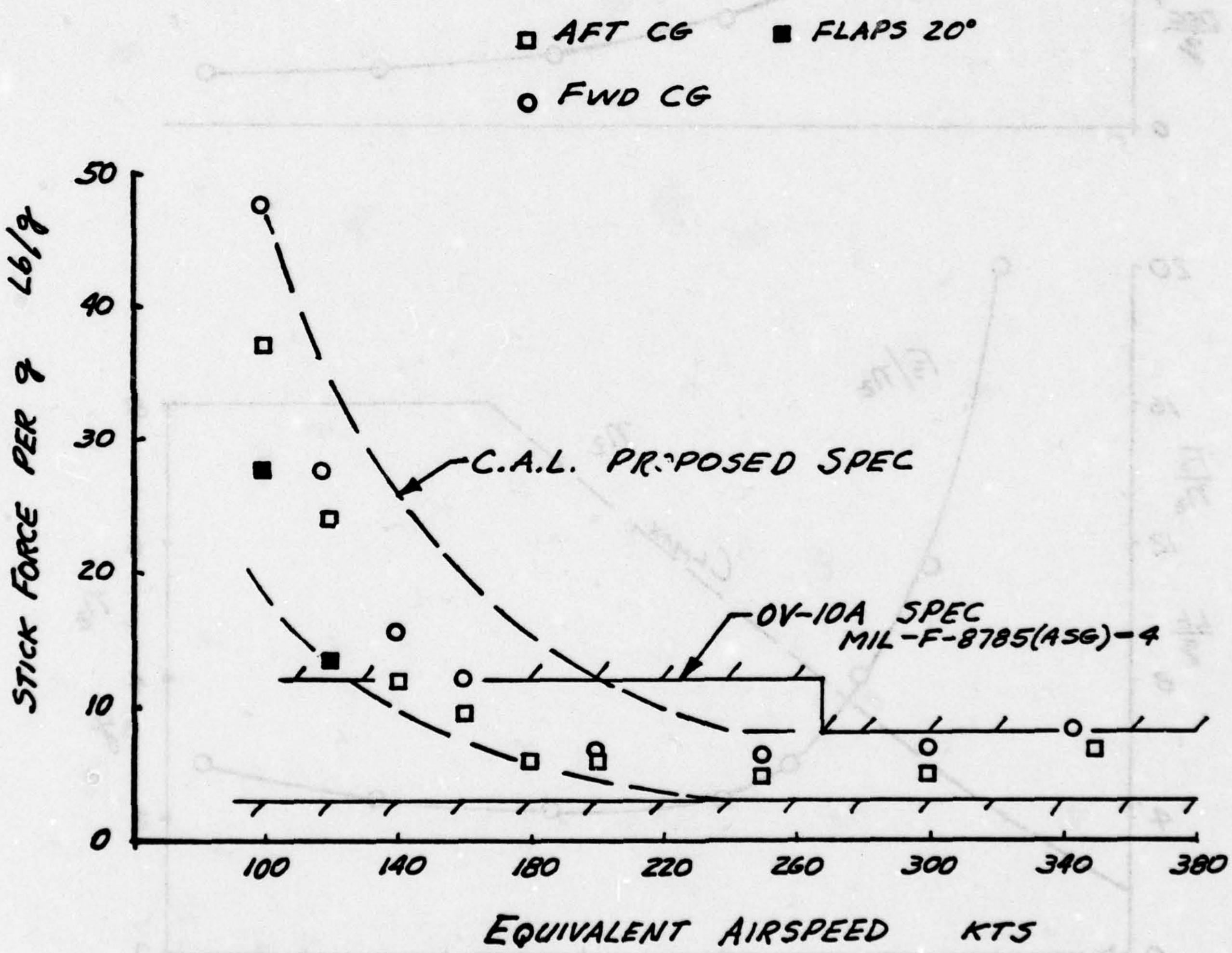

FIG. 3 OV-10A STICK FORCE PER G

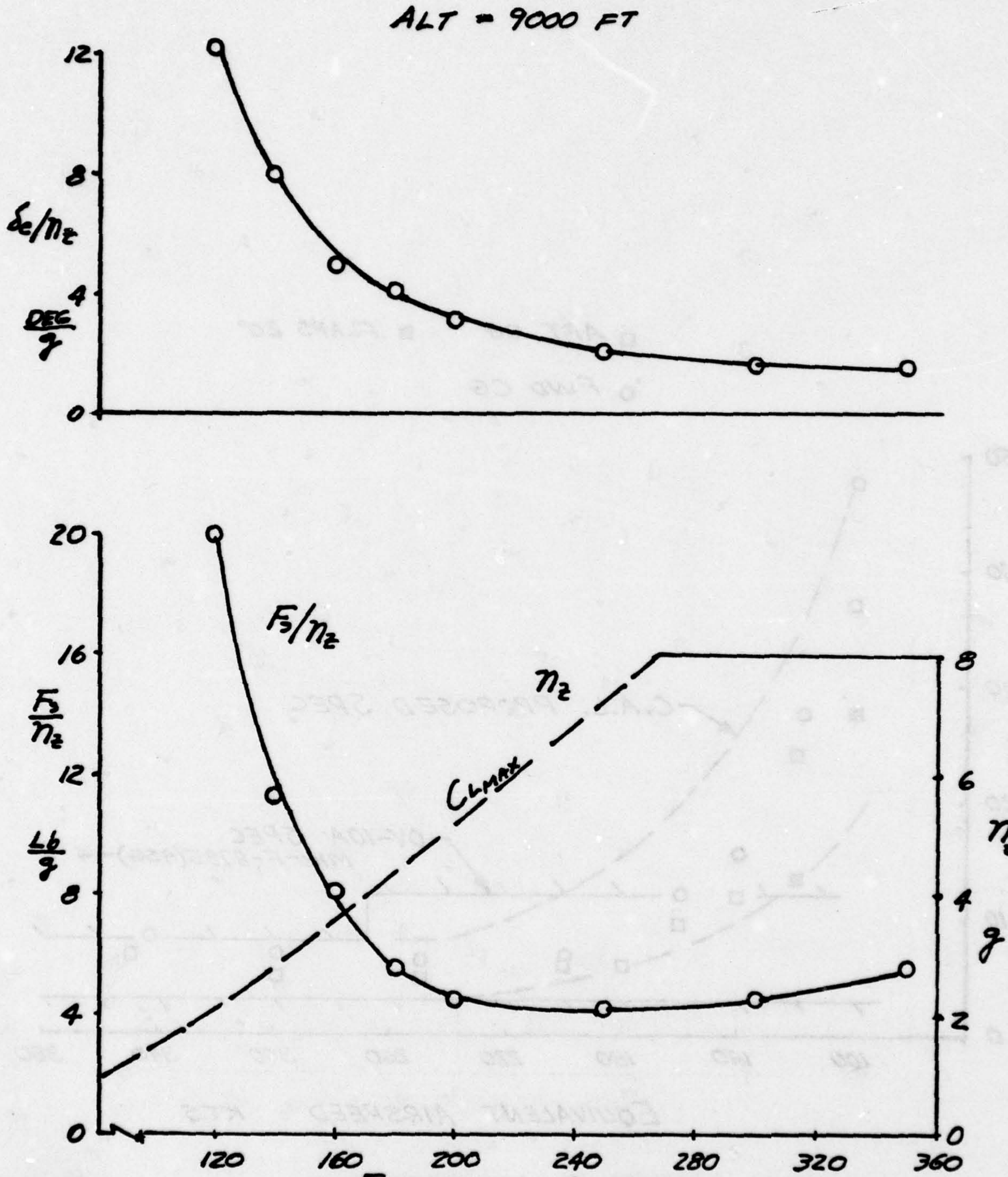

FIG. 4 OV-10A MANEUVERING CONTROL

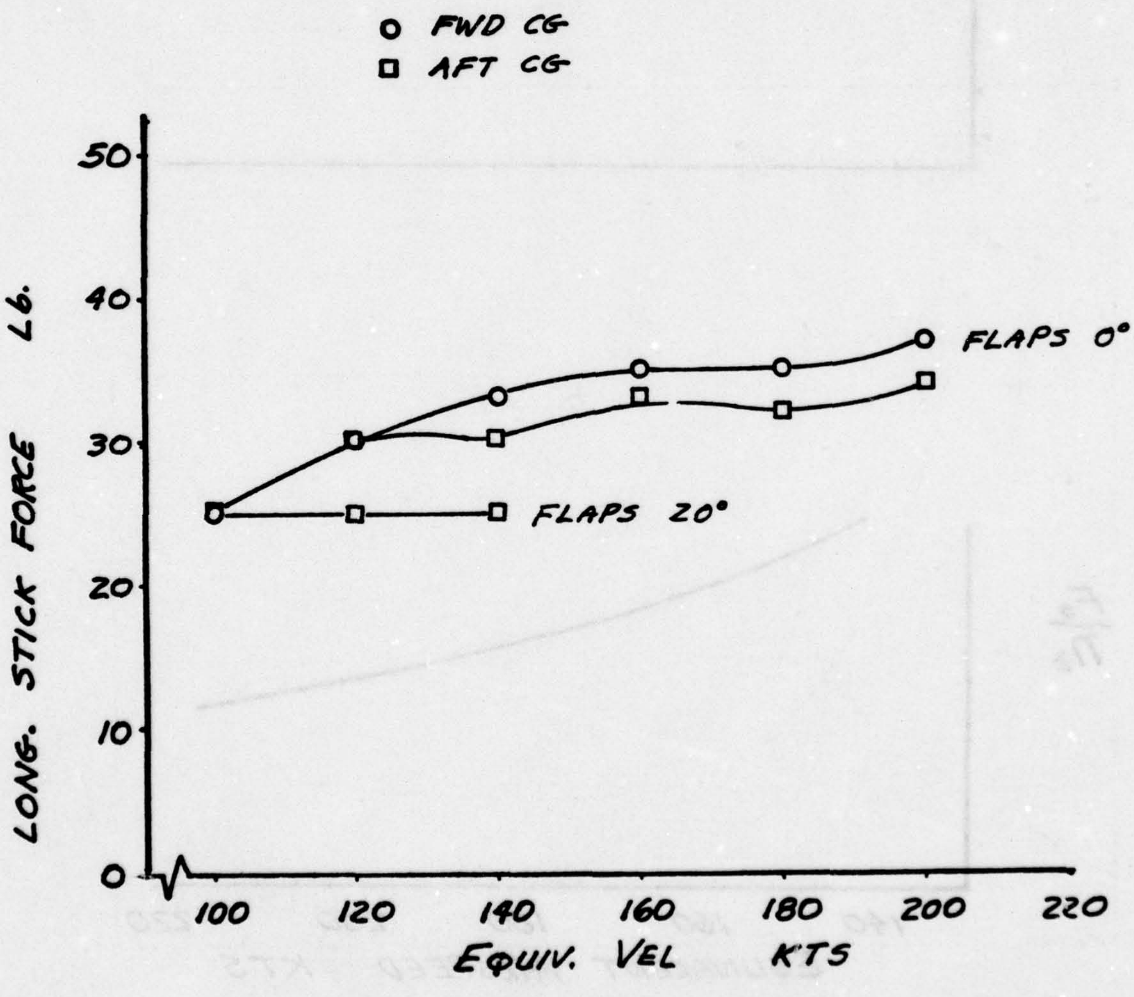

FIG. 5 LONG. STICK FORCE AT STALL

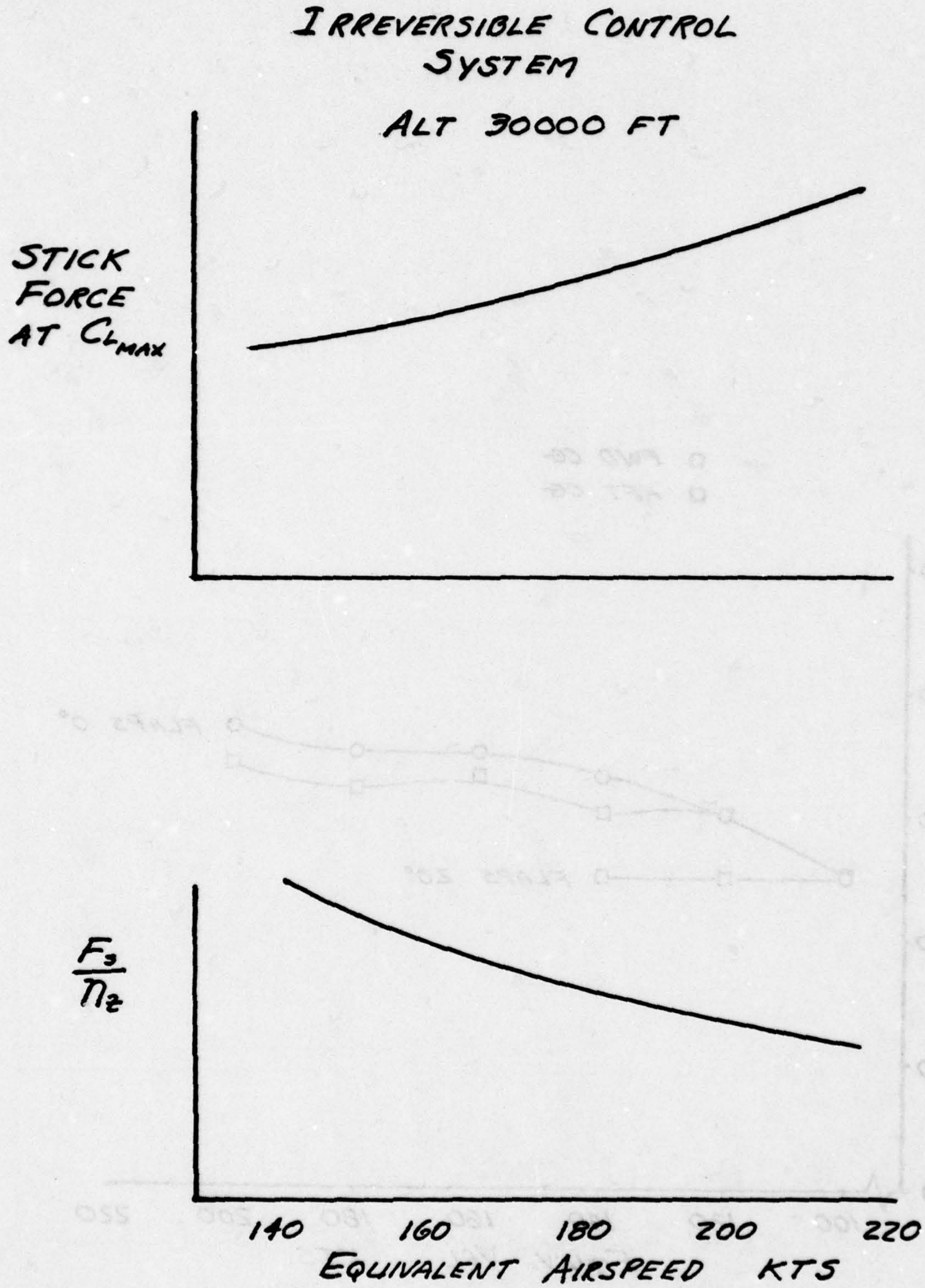

FIG. 6 IRREVERIBLE CONTROL FORCE

Dwight Schaeffer, Boeing: Do you feel force at limit load factor (or max load factor) is all that is important? Are you also concerned with forces to exceed n_L being too light?
Answer: The minimum force gradients specified in the spec should still apply at low speeds as well. I think the max stick force at limit lift is more important than the max force gradient at low speeds.

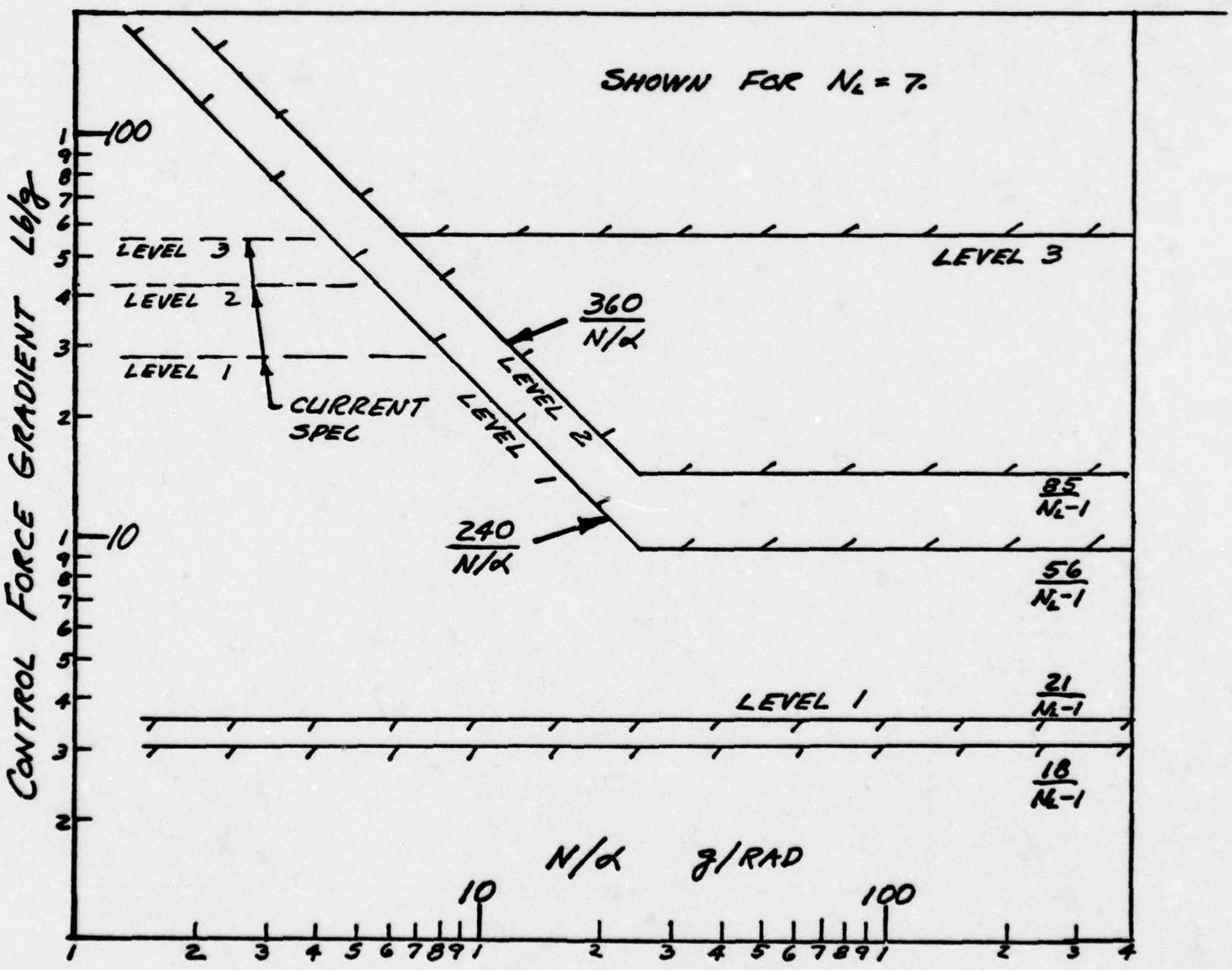

FIG. 7 FORCE GRADIENT REQUIREMENTS

Dwight Schaeffer, Boeing: Do you feel force at limit load factor (or max load factor) is all that is important? Are you also concerned with forces to exceed n_z being too light?
Answer: The minimum force gradients specified in the spec should still apply at low speeds as well. I think the max stick force at limit lift is more important than the max force gradient at low speeds.

A SUGGESTED CRITERION FOR PILOTED HEADING CONTROL

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NOMENCLATURE

g	Acceleration due to gravity
L	Aerodynamic roll moment divided by roll moment of inertia
L_λ	$\partial L/\partial\lambda$ where $\lambda = \delta_w$, φ, or β
L'_λ	$[L_\lambda + (I_{xz}/I_x)N_\lambda]/[1 - I_{xz}^2/I_xI_z)]$
N	Aerodynamic yawing moment divided by yaw moment of inertia
N_λ	$\partial N/\partial\lambda$ where $\lambda = \delta_w$, δ_r, β, r, or p
N'_λ	$[N_\lambda + (I_{xz}/I_z)L_\lambda]/[1 - (I_{xz}^2/I_xI_z)]$
N^λ_δ	Numerator of λ/δ transfer function
p	Roll rate
r	Yaw rate
s	Laplace operator
U_o	Steady-state velocity
Y_{CF}	Aileron-to-rudder crossfeed function
Y'_{CF}	See Equation 6
Y_{δ_α}	$\partial Y/\partial\delta_\alpha$
β	Sideslip angle
$\Delta\beta_{max}$	Maximum sideslip excursion at the c.g. occurring within two seconds or one-half period of the dutch roll, whichever is greater, for a step aileron control command (see Ref. 1)
δ_r	Rudder pedal deflection at the cockpit
$\delta_{r(3)}$	Rudder pedal deflection 3 seconds after a unit step lateral wheel input
$\delta'_{r(3)}$	Normalized rudder deflection; $\delta'_{r(3)} = N_{\delta_r}\delta_{r(3)}$
δ_w	Lateral wheel (or stick) deflection at the cockpit

Δ	Characteristic determinant-denominator for transfer functions
ζ_d	Dutch roll damping
μ	Crossfeed shaping parameter (See Equation 10)
φ	Bank angle
$\lvert\varphi/\beta\rvert_d$	Roll/sideslip ratio in the dutch roll mode
$\varphi_{osc}/\varphi_{ave}$	A measure of the ratio of the oscillatory component of bank angle to the average component of bank angle following a rudder-pedals free impulse aileron control command (see Ref. 1)
ψ_β	Phase angle expressed as a lag for a cosine representation of the dutch roll oscillation in sideslip (Ref. 1)
ω_d	Dutch roll frequency

INTRODUCTION

The ability to make precise changes in aircraft heading is a key factor in pilot evaluation of lateral-directional handling qualities. Assuming other good qualities (e.g., adequate roll response, yaw frequency/damping, etc., per Ref. 1), deficiencies in heading control, which can nevertheless exist, are directly traceable to excitation of the dutch roll mode due to roll-yaw crosscoupling effects. It is commonly accepted piloting technique to reduce these excursions by appropriate use of the aileron and rudder, usually referred to as "coordinating the turn." The problem is that existing criteria (see, for instance, Refs. 1-4) for heading control are based on aileron-only parameters, and the effects of rudder control are only indirectly apparent as they may have influenced individual pilot ratings. The fact that these criteria are not satisfactory is shown in Ref. 5, where several configurations which violated boundaries based on aileron-only parameters were given good to excellent pilot ratings. The approach taken here is that for an otherwise acceptable airplane the aileron-rudder shaping necessary to coordinate the turn is a dominant factor in pilot evaluation of heading control. In this regard it is important to recognize that heading control is basically an outer loop and cannot be satisfactory if the inner bank angle loop is unsatisfactory. Table 1 contains a set of requirements intended to serve as a checklist for good roll control. These requirements are discussed in some detail in Ref. 6.

TABLE 1

GROUND RULES FOR APPLICATION OF RATING DATA TO HEADING CONTROL CRITERIA

1) $T_R < 1.25$

2) $\omega_d > 0.4$

3) $\zeta_d > 0.08$ and $\zeta_d\omega_d > 0.15$

4) $|\varphi/\beta|_d < 1.5$ when turbulence is a factor and $|N'_{\delta_w}/L'_{\delta_w}| > 0.03$

5) Meets L'_β vs. ω_d boundaries when $|N'_{\delta_w}/L'_{\delta_w}| \leq 0.03$

6) Meets Level 2 $\varphi_{osc}/\varphi_{ave}$ in MIL-F-8785B

7) Pilot comments do not indicate:
   a) Significant roll control problems
   b) Control power or sensitivity problems
   c) Nonlinear control system problems such as friction, breakout, etc.
   d) Excessive gust response

ANALYSIS AND BASIC CONCEPT

In general, coordinated flight implies minimum yaw coupling due to roll entries and exits which can be quantified in many ways, e.g.: 1) zero sideslip angle ($\beta = 0$); 2) zero lateral acceleration at the c.g.; 3) turn rate consistent with bank angle and speed ($r = g\varphi/U_o$); and 4) zero lateral acceleration at the cockpit (ball in the middle).

Conditions 1-3 are equivalent when the side force due to Y_{δ_a} and Y_r are very small, which is usually the case. The fourth turn coordination criterion is complicated by pilot location effects which, however, appear to be mainly associated with ride qualities and not with heading control itself (Ref. 5). Based on these considerations it appears that sideslip angle is an appropriate indicator of turn coordination. Accordingly, the following formulation undertakes to identify the parameters that govern the aileron-rudder shaping required to maintain coordinated flight as defined by zero sideslip angle ($\beta = 0$).

With an aileron-rudder crossfeed, Y_{CF}, the rudder by definition, is given by

$$\delta_r \equiv Y_{CF}\,\delta_w \tag{1}$$

For the assumed ideal (zero sideslip) coordination

$$\beta = \left(\frac{N^{\beta}_{\delta_w}}{\Delta} + Y_{CF}\frac{N^{\beta}_{\delta_r}}{\Delta}\right)\delta_w \equiv 0 \tag{2}$$

whereby the ideal crossfeed is

$$Y_{CF} \equiv \frac{\delta_r}{\delta_w} = -\frac{N^{\beta}_{\delta_w}}{N^{\beta}_{\delta_r}} \tag{3}$$

For augmented airplanes, these numerators are high order and cannot be generalized. However, as was shown in Ref. 7, aircraft with complex augmentation systems represented by higher-order systems (HOS) tend to respond to pilot inputs in a fashion similar to conventional unaugmented aircraft or low-order systems (LOS). In fact, more recent unpublished work by the author of Ref. 7 showed that a HOS which *cannot* be fit to a *LOS* form is predictably unsatisfactory to the human pilot.

The appropriate LOS form for Y_{CF} is based on the approximate factors for conventional airplanes obtained from Ref. 8 as follows.

$$Y_{CF} = \frac{N'_{\delta_w}[s + A_w(g/U_o)][s + (1/T_{\beta_w})]}{Y^{*}_{\delta_r}[s + A_r(g/U_o)][s + (1/T_{\beta_r})][s - (N'_{\delta_r}/Y^{*}_{\delta_r})]} \tag{4}$$

where

$$A_i = \frac{L'_r - (L'_{\delta_i}/N'_{\delta_i})N'_r}{L'_p - (L'_{\delta_i}/N'_{\delta_i})[N'_p - (g/U_o)]}$$

$$i = w \text{ or } r$$

$$1/T_{\beta_i} = -L'_p + (L'_{\delta_i}/N'_{\delta_i})[N'_p - (g/U_o)]$$

For the frequency range of interest, i.e., excluding both low and high frequencies $[A_i(g/U_o) << s << N'_{\delta_a}/Y^*_{\delta_r}]$

$$Y_{CF} \doteq -\frac{N'_{\delta_w}[s + (1/T_{\beta_w})]}{N'_{\delta_r}[s + (1/T_{\beta_r})]} \tag{5}$$

To provide a meaningful reference for the control crosscoupling term, N'_{δ_w}, in Eq. 5, it is expressed as the ratio of yawing to rolling acceleration, $N'_{\delta_w}/L'_{\delta_w}$. Also, since the rudder sensitivity can be separately optimized and does not usually represent a basic airframe limitation, it is appropriate to remove it from consideration. Accordingly, the resulting LOS representation of the crossfeed, Y'_{CF}, is given as*:

$$Y'_{CF} \equiv Y_{CF}\frac{N'_{\delta_r}}{L'_{\delta_w}} = -\frac{N'_{\delta_r}}{L'_{\delta_w}}\frac{N^{\beta}_{\delta_w}}{N^{\beta}_{\delta_r}} \doteq -\frac{N'_{\delta_w}[s + (1/T_{\beta_w})]}{L'_{\delta_w}[s + (1/T_{\beta_r})]} \tag{6}$$

Equation 6 indicates that the aileron-to-rudder shaping required to maintain coordinated flight ($\beta = 0$) is directly related to the separation between the aileron (wheel or stick) and rudder (pedal) sideslip zeros.

As a basis for direct correlation with pilot opinion, a "rudder shaping parameter," μ, is defined in "theoretical" form as

$$\mu = (T_{\beta_r}/T_{\beta_w}) - 1 \tag{7}$$

The frequency response characteristics of Y'_{CF}, Eq. 6, as function of the sign of μ are shown in Fig. 1 in terms of literal expressions for the Bode asymptotes. These asymptotes indicate that the magnitude of the coordinating rudder is a function of $N'_{\delta_w}/L'_{\delta_w}$ at all frequencies and that the shaping of the rudder response is determined by μ. These parameters are summarized in terms of their analytical and pilot-centered functions in Table 2.

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*All derivatives are in the stability axis system.

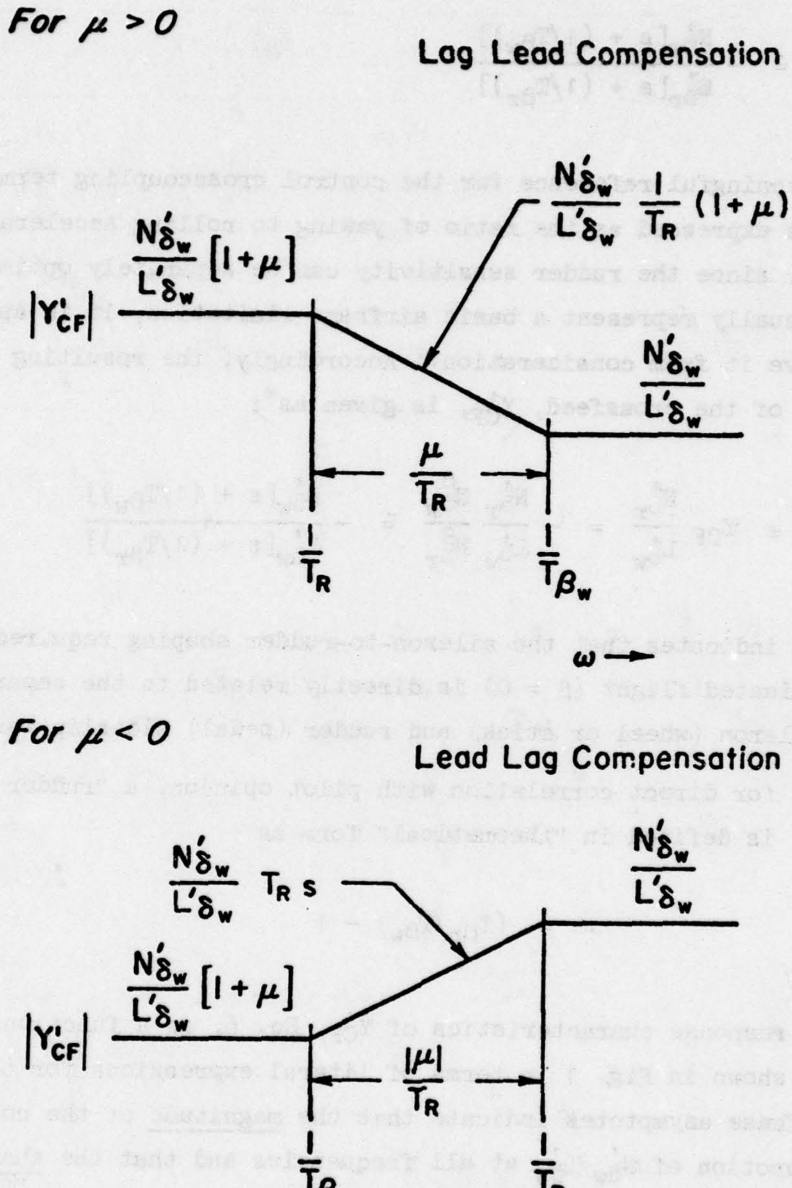

Figure 1. Asymptotes of Aileron-Rudder Crossfeed

TABLE 2

PARAMETERS DEFINING THE LOS REPRESENTATION OF THE AILERON-RUDDER CROSSFEED

PARAMETER	ANALYTICAL FUNCTION	PILOT-CENTERED FUNCTION
μ	Defines shape of Y'_{CF}	Determines complexity of rudder activity necessary for ideally coordinated turns. Also defines phasing of heading response when rudder is not used.
$N'_{\delta_w}/L'_{\delta_w}$	Defines magnitude of Y'_{CF}	Determines magnitude of rudder required and/or high-frequency yawing induced by aileron inputs.

The parameters $N'_{\delta_w}/L'_{\delta_w}$ and μ are a natural choice for correlation of heading control pilot rating data since they completely define the aileron-to-rudder crossfeed necessary for turn coordination. Such an ideal crossfeed is difficult to isolate with simple flight test procedures, but is nevertheless considered a viable correlation concept because of modern usage (Ref. 1) which permits simulation and analysis methods to demonstrate specification compliance.

Since the rudder <u>sequencing</u> with aileron inputs is the key issue, it was decided to use a LOS form in the time domain. Assuming a unity high-frequency gain for the Eq. 6 form, and the ideal definition of μ (Eq. 7), the rudder time history required to coordinate a unit step wheel or stick input is:

$$\delta_r(t) = 1 + \mu(1 - e^{-t/T_{\beta r}}) \tag{8}$$

Note that $\delta_r(t)$ refers to the rudder <u>pedal</u> motion (thereby including effects of rudder gearing and accounting for the SAS). Solving Eq. 8 for the rudder shaping parameter, μ:

$$\mu = \frac{\delta_r(t) - 1}{1 - e^{-t/T_{\beta r}}} \tag{9}$$

The value of t used is properly set by the lower limit on the frequency range of interest for piloted heading control. The simulation experiments of Ref. 9 indicated that a minimum heading crossover of about 1/3 rad/sec was necessary for desirable handling qualities. Therefore, a corresponding time of 3 sec was selected as being most pertinent to a pilot-centered characterization of crossfeed properties. Recognizing further (Eq. 4) that $T_{\beta_r} \doteq -1/L_p'$ is approximately equal to the roll mode time constant, T_R, and that the latter must generally be less than 1.0 to 1.4 sec for acceptable roll control (Ref. 1) sets the following limits on the exponential in Eq. 9.

$$T_R \le 1.0^* \qquad e^{-3/T_R} \le 0.049$$
$$\le 1.4^\dagger \qquad \le 0.117$$

Accordingly, Eq. 9 reduces within a maximum error of 5-10 percent, depending on airplane class, to

$$\mu \doteq \delta_r(3) - 1 \tag{10}$$

This simple relationship was used to compute μ for the pilot rating correlations later shown.

However, before this simple formula can be applied it is necessary to avoid the high-frequency responses which occur due to pairs of roots which frequently occur with complex SAS installations having associated higher-order β numerators. For example, a simple washed out yaw rate feedback and a first-order lagged aileron rudder crossfeed results in seventh-order β numerators of unaugmented airplanes. Most of the zeros of these polynomials occur at very high frequency, having negligible effect on the dynamics near the pilot's crossover frequency, and therefore should not be accounted for in the shaping function μ. The standard procedure utilized to compute the values of μ was to eliminate all roots of the β numerators above values of 6 rad/sec in pairs, i.e., keeping their order relative to each other the

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*For small, light, or highly maneuverable airplanes.

†For medium to heavy weight, low to medium maneuverability airplanes.

same (e.g., a third over fourth would be reduced to a second over third order, etc.). Roots above 6 rad/sec which do not occur in pairs are left unmodified.

The following example illustrates a typical computation of μ and the effect of removing the high-frequency roots from Eq. 2. The aileron-rudder crossfeed for one of the Ref. 10 configurations used in the pilot rating correlations is given as:

$$\frac{\delta_r}{\delta_w} = \frac{.19(s - .102)(s - .922)(s + 605.2)}{(s - .057)(s + 5.6)(s + 109.9)} \tag{11}$$

As discussed above, all roots above 6 rad/sec are removed in pairs and the high-frequency gain is set to unity, resulting in the following equation:

$$\frac{\delta_r}{\delta_w} = \frac{(s + .102)(s - .922)}{(s - .057)(s + 5.6)} \tag{12}$$

The rudder time responses to a unit wheel input for Eqs. 11 and 12 are plotted in Fig. 2. Removal of the high-frequency roots is seen to replace

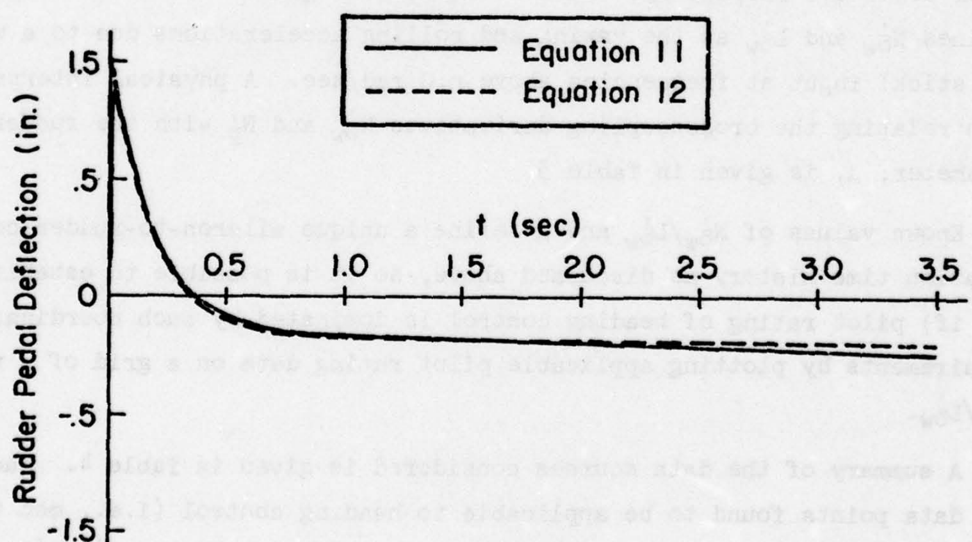


Figure 2. Effect of Removing High-Frequency Roots from β Numerators

the initial rapid rudder reversal with a unity initial condition. These responses are essentially equivalent to the pilot who sees the necessity to use immediate rudder with aileron inputs (which must be removed 1/2 sec later). The value of μ corresponding to this response is $\delta_r(3) - 1 = -1.17$.

Figure 3 presents typical coordinating ($\beta = 0$) rudder time histories for step aileron inputs on a grid of μ vs. $N'_{\delta_w}/L'_{\delta_w}$. Moving vertically on this grid changes the shape (μ) of the crossfeed, Y_{CF}, keeping the initial value (high-frequency gain) constant. Moving horizontally produces a change in the crossfeed gain ($N'_{\delta_w}/L'_{\delta_w}$) at all frequencies without changing the shape. Note that this is consistent with Table 2 and Fig. 1, where it is shown that μ dictates the required aileron-to-rudder shaping and N'_{δ_w} defines the magnitude of the gain for all times (and frequencies). The basic shapes of the time histories in Fig. 3 are indicative of the fundamental assumption that the rudder time history can be fit by the Eq. 6 form. The basic implication of this form is that the rudder response is essentially monotonic in the frequency range of interest.

For augmented airplanes the effective values of $N'_{\delta_w}/L'_{\delta_w}$ (which represent the high-frequency yawing and rolling accelerations) are taken as the high-frequency gain of the simplified β/δ_w and φ/δ_w transfer functions, e.g., all roots above 6.0 rad/sec are taken as equivalent gains. In effect this defines N'_{δ_w} and L'_{δ_w} as the yawing and rolling accelerations due to a wheel (or stick) input at frequencies above 6.0 rad/sec. A physical interpretation relating the crosscoupling derivatives N'_{δ_w} and N'_p with the rudder shaping parameter, μ, is given in Table 3.

Known values of $N'_{\delta_w}/L'_{\delta_w}$ and μ define a unique aileron-to-rudder coordination time history as discussed above, so it is possible to establish how (or if) pilot rating of heading control is dominated by such coordination requirements by plotting applicable pilot rating data on a grid of μ vs. $N'_{\delta_w}/L'_{\delta_w}$.

A summary of the data sources considered is given in Table 4. Each of the data points found to be applicable to heading control (i.e., met the ground rules) is plotted and faired on a logarithmic grid of $N'_{\delta_w}/L'_{\delta_w}$ vs. μ in Fig. 4. Only in-flight and moving-base simulator data were considered.

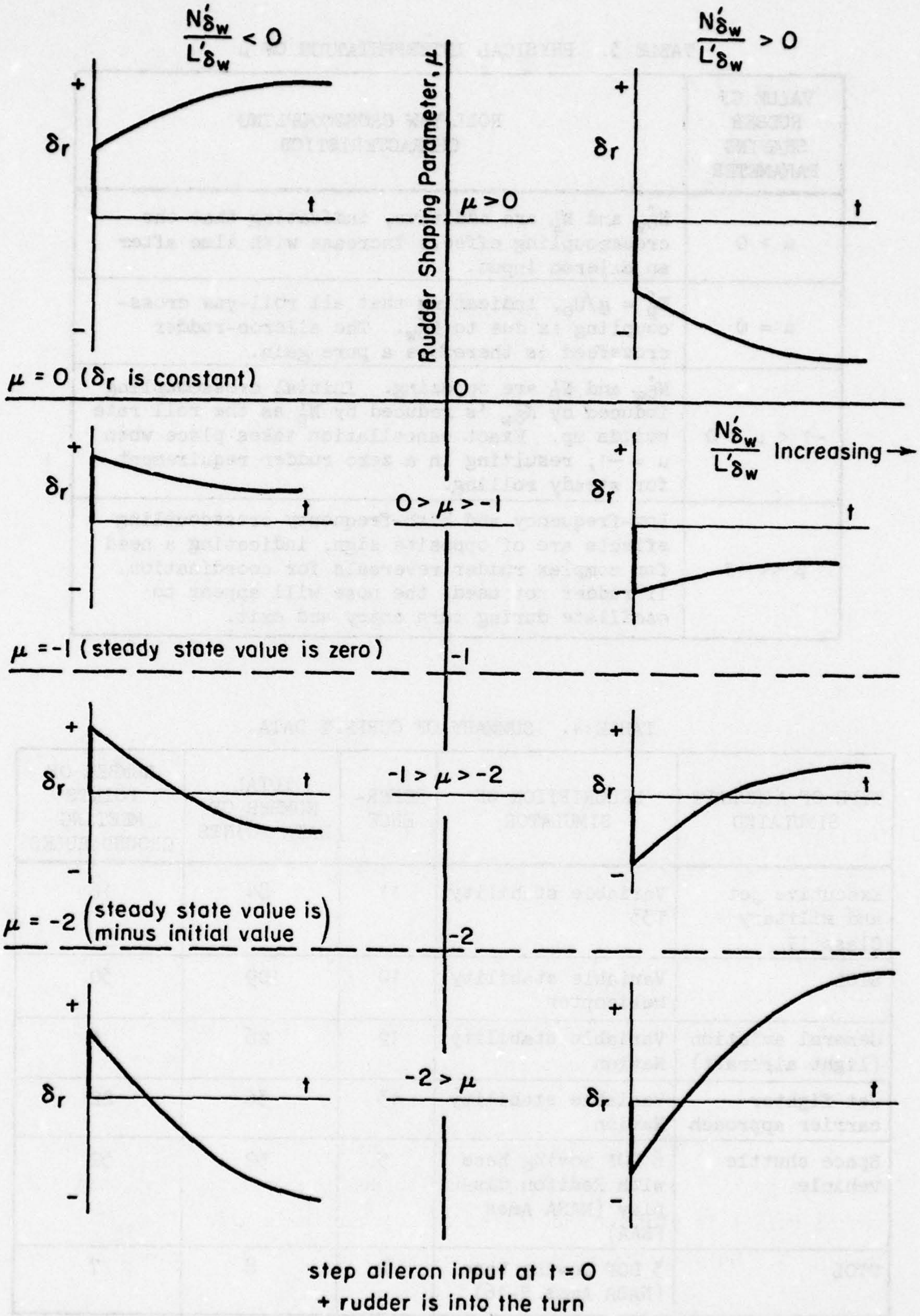


Figure 3. Typical Rudder Time Histories for Zero Sideslip

TABLE 3. PHYSICAL INTERPRETATION OF μ

VALUE OF RUDDER SHAPING PARAMETER	ROLL-YAW CROSSCOUPLING CHARACTERISTICS
$\mu > 0$	N'_{δ_w} and N'_p are additive, indicating that the crosscoupling effects increase with time after an aileron input.
$\mu = 0$	$N'_p = g/U_o$, indicating that all roll-yaw crosscoupling is due to N'_{δ_w}. The aileron-rudder crossfeed is therefore a pure gain.
$-1 < \mu < 0$	N'_{δ_w} and N'_p are opposing. Initial crosscoupling induced by N'_{δ_w} is reduced by N'_p as the roll rate builds up. Exact cancellation takes place when $\mu = -1$, resulting in a zero rudder requirement for steady rolling.
$\mu \ll -1$	Low-frequency and high-frequency crosscoupling effects are of opposite sign, indicating a need for complex rudder reversals for coordination. If rudder not used, the nose will appear to oscillate during turn entry and exit.

TABLE 4. SUMMARY OF CURRENT DATA

TYPE OF AIRCRAFT SIMULATED	DESCRIPTION OF SIMULATOR	REFERENCE	TOTAL NUMBER OF DATA POINTS	NUMBER OF POINTS MEETING GROUND RULES
Executive jet and military Class II	Variable stability T33	11	84	16
STOL	Variable stability helicopter	10	109	30
General aviation (light aircraft)	Variable stability Navion	12	26	6
Jet fighter-carrier approach	Variable stability Navion	13	36	22
Space shuttle vehicle	6 DOF moving base with Redifon display (NASA Ames FSAA)	5	52	52
STOL	3 DOF moving base (NASA Ames S-16)	2	8	7

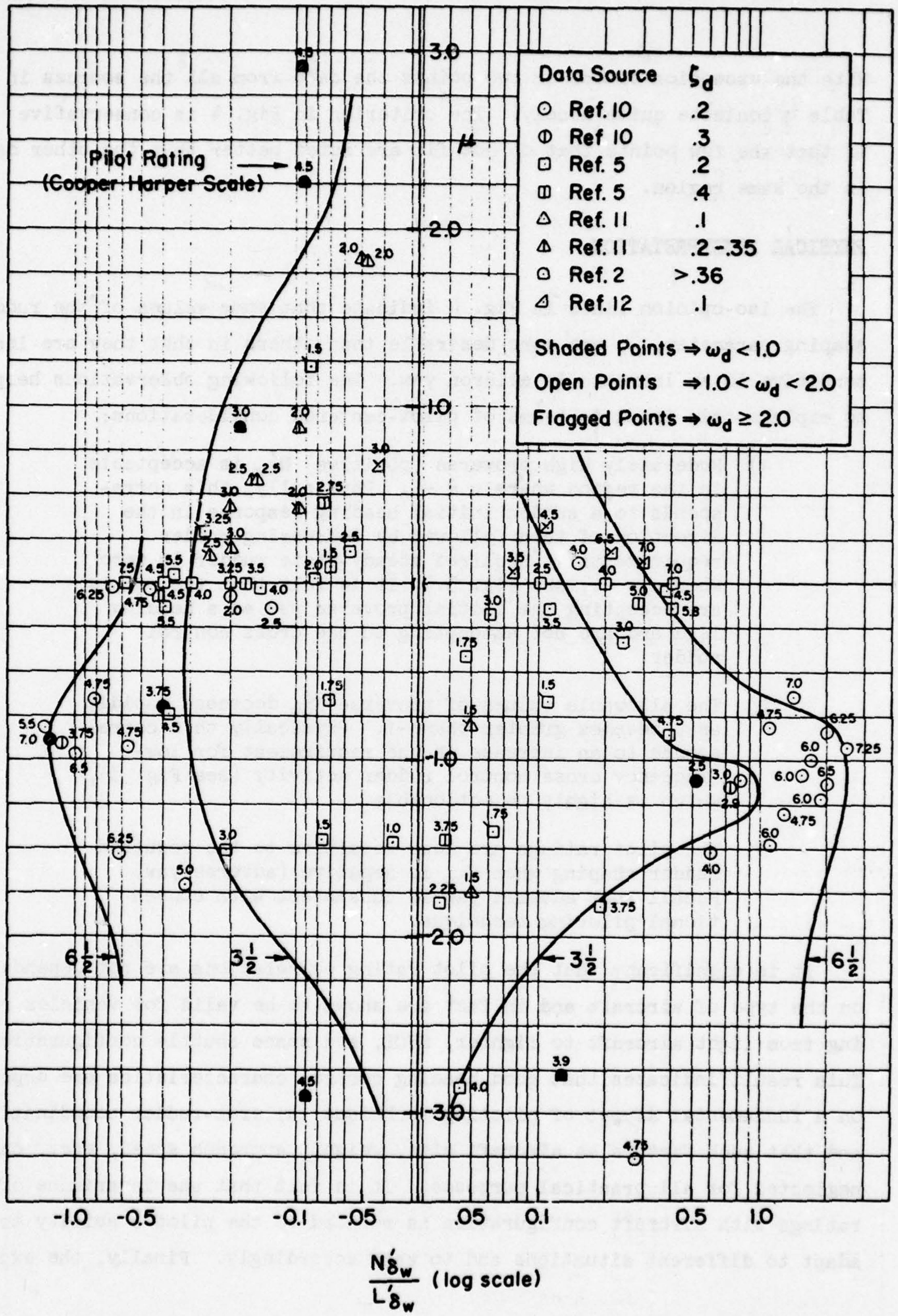


Figure 4. Pilot Rating Correlation with Crossfeed Parameters

With the exception of one or two points the data from all the sources in Table 4 coalesce quite nicely. The criterion in Fig. 4 is conservative in that the few points that do not fit are rated better than the other data in the same region.

PHYSICAL INTERPRETATION

The iso-opinion lines in Fig. 4 indicate that some values of the rudder shaping parameter, μ, are more desirable than others in that they are less sensitive to an increase in aileron yaw. The following observations help to explain this trend in terms of pilot-centered considerations:

1) Moderately high proverse (positive) N'_{δ_w} is acceptable in the region where $\mu \doteq -1$. Physically, this corresponds to a sudden initial heading response in the direction of turn followed by decreasing rudder requirements. (Required steady-state rudder is zero when $\mu = -1$, see Fig. 3.) It is felt that the pilots are accepting the initial proverse yaw as a heading lead and are not attempting to use cross control rudder.

2) The allowable values of proverse N'_{δ_w} decrease rapidly as μ becomes greater than -1. Physically this corresponds to an increase in the requirement for low-frequency cross control rudder activity (see Fig. 3), which is highly objectionable.

3) The pilot ratings are less sensitive to the required rudder shaping when N'_{δ_w} is negative (adverse yaw). Recall that adverse yaw is consistent with conventional piloting technique.

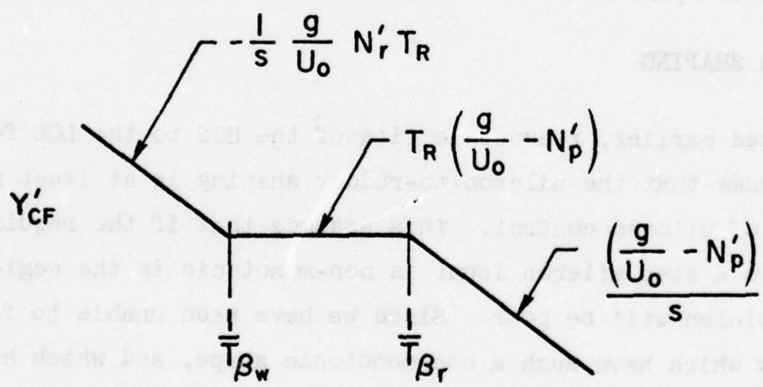
It is significant that the pilot rating correlations are not dependent on the type of aircraft and in fact are shown to be valid for vehicles ranging from light aircraft to fighter, STOL, and space shuttle configurations. This result indicates that good heading control characteristics are dependent on a fundamental aspect of piloting technique (aileron-rudder coordination) and that such factors as aircraft size, weight, approach speed, etc., can be neglected for all practical purposes. It is felt that the invariance of ratings with aircraft configuration is related to the pilot's ability to adapt to different situations and to rate accordingly. Finally, the excellent

correlations of pilot ratings with the aileron-rudder crossfeed characteristics indicates that the required rudder coordination is indeed a dominant factor in pilot evaluation of heading control.

The rudder shaping parameter is attractive as a heading control criterion because the handling quality boundaries are easily interpreted in terms of pilot-centered considerations.

$N'_{\delta_w}/L'_{\delta_w}$ NEAR ZERO

Control crosscoupling effects are obviously not a factor when $|N'_{\delta_w}/L'_{\delta_w}|$ is small. This may occur when the basic control crosscoupling is negligible or with augmentation systems which result in ideal crossfeeds, Y_{CF}, having denominators of higher-order dynamics than numerators (e.g., the augmented N'_{δ_w} is zero). For $N'_{\delta_w}/L'_{\delta_w}$ identically zero, the required aileron-rudder crossfeed takes the Bode asymptote form shown in Fig. 5 for unaugmented conventional airplanes. The rudder magnitude required to coordinate mid-frequency and high-frequency aileron (wheel) inputs is seen to be dependent on the roll crosscoupling, $g/U_o - N'_p$, whereas low-frequency rudder requirements are dependent on N'_r. The required rudder shaping has the characteristics of a rate system (ramp δ_r to step δ_w input) at low and high frequency.



$$\frac{1}{T_{\beta_w}} \doteq \frac{-\frac{g}{U_o}N'_r}{\left(\frac{g}{U_o} - N'_p\right)}$$

Figure 5. Required Crossfeed for $N'_{\delta_w} = 0$

Accordingly, aileron-rudder shaping per se is not the essence of the problem, which reduces, instead, to concern with the general magnitude of the required rudder crossfeed. Utilizing the same response considerations as in the computation of μ, $\delta_r(3)$ is suggested as the correlating parameter when $|N'_{\delta_w}/L'_{\delta_w}|$ is small or when the denominator of Y_{CF} is of higher order than the numerator. In order to remove rudder sensitivity effects, which can be separately optimized, $\delta'_r(3) = N'_{\delta_r}\delta_r(3)$ is used as a correlating parameter. The question of what specifically constitutes a "small" value of $N'_{\delta_w}/L'_{\delta_w}$ has proven to be somewhat difficult to quantify. Reasonably good correlations were found by plotting the $N'_{\delta_w}/L'_{\delta_w} < 0.03$ pilot ratings vs. $\delta'_r(3)$ as shown in Fig. 6. More recent experience in utilizing the μ parameter has revealed several unacceptable configurations, where $N'_{\delta_w}/L'_{\delta_w}$ is slightly greater than 0.03, that fall within the acceptable region in Fig. 4. Plotting these configurations on the $\delta'_r(3)$ criterion revealed the deficiency in all cases tried.

Based on this experience, the rules for application of the μ parameter have been revised as follows. If $|N'_{\delta_w}/L'_{\delta_w}| < 0.07$, plot μ vs. $N'_{\delta_w}/L'_{\delta_w}$ on the Fig. 4 criterion and $\delta'_r(3)$ on the Fig. 6 criterion and utilize the most conservative result. Note that $\delta'_r(3)$ is simply the $\delta_r(3)$ used in the μ calculation (Eq. 10) multiplied by L'_{δ_w} [recall that $Y'_{CF} = Y_{CF}(N'_{\delta_w}/L'_{\delta_w})$ and that $\delta'_r(3) = N'_{\delta_r}\delta_r(3)$]. If $|N'_{\delta_w}/L'_{\delta_w}| > 0.07$ the Fig. 4 criterion is used without the need to check $\delta'_r(3)$.

COMPLEX RUDDER SHAPING

As discussed earlier, reasonable fits of the HOS to the LOS form implicit in Eq. 10 presume that the aileron-to-rudder shaping is at least monotonic in the region of piloted control. This assumes that if the required rudder coordination to a step aileron input is non-monotonic in the region of control, pilot opinion will be poor. Since we have been unable to find any configurations which have such a non-monotonic shape, and which have been tested for pilot opinion of heading control, it is not possible to quantify the mismatch effects at this time.

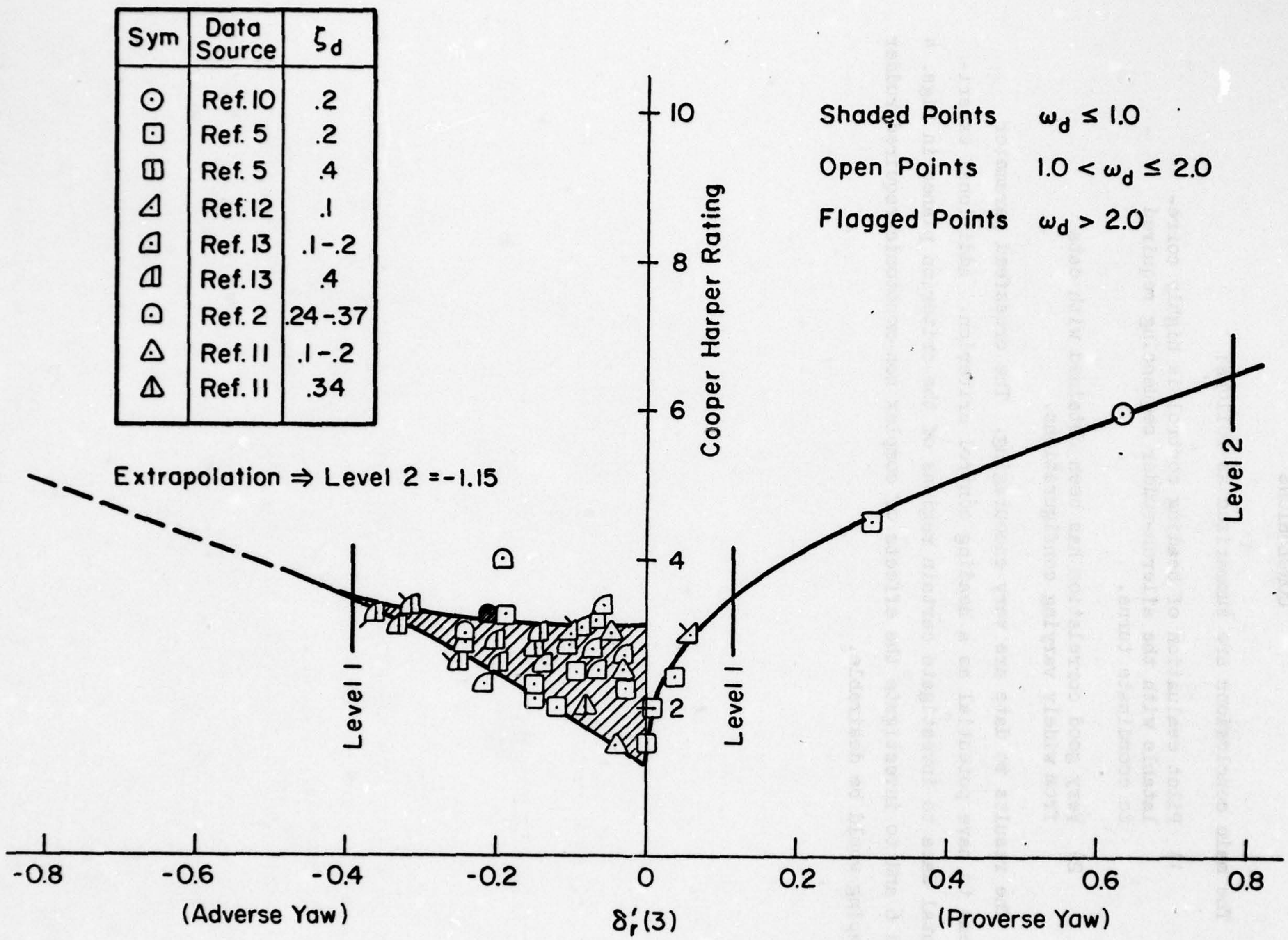


Figure 6. Pilot Rating Correlations When $|N'_{\delta_w}/L'_{\delta_w}|$ Is Small

CONCLUSIONS

The main conclusions are summarized as follows:

1) Pilot evaluation of heading control is highly correlatable with the aileron-rudder sequencing required to coordinate turns.

2) Very good correlation has been obtained with data from widely varying configurations.

The results to date are very encouraging. The crossfeed parameter seems to have potential as a heading control criterion. Additional experimental data to investigate certain regions of the criterion planes in Figs. 4 and 6 and to investigate the effects of complex non-monotonic required rudder shaping would be desirable.

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APPENDIX

In the Calspan critique (Ref. 14), two example configurations are used to question the viability of the μ parameter. This appendix is presented not only as a rebuttal, but also to illustrate some examples of proper application of the parameter.

The first example is configuration P8 from Ref. 15. It was specifically chosen (in Ref. 14) because the pilot ratings and flight test commentary indicated severe deficiencies in heading control (average rating = 8 and maximum rating = 10, with three pilots), yet it plots inside the Level 2 boundary on the current $\Delta\beta/k$ specification. The configuration was plotted on the μ criterion boundaries of Fig. 4 and shown to fall within the Level 1 region. Unfortunately, it was plotted incorrectly in that $N'_{\delta_W}/L'_{\delta_W}$ was not converted to stability axis. More important, however, is the fact that for low values of control crosscoupling such as for Configuration P8, the $\delta'_r(3)$ parameter should also be calculated. Since $\delta'_r(3)$ is simply $\delta_r(3) \times L'_{\delta_W}$, it can be obtained directly from Ref. 14, Fig. 3 (which plots δ_r vs. time) as $-1.7 \times 1.04 = -1.77$ ($L'_{\delta_W} = 1.04$ was obtained from Ref. 15). From Fig. 6 it can be seen that a $\delta'_r(3)$ of -1.77 represents such extreme adverse yaw that it does not even fit on the plot! However, if we extend the linear extrapolation, a predicted rating of 8.5 results, which is in excellent agreement with the pilot ratings. It should be noted that in the strictest sense the version of μ published in Ref. 6 indicated that $\delta'_r(3)$ applied when $|N'_{\delta_W}/L'_{\delta_W}|$ is less than 0.03, whereas a value of 0.035 was utilized in Ref. 14. However, it should have been realized that even a slight reduction in airspeed, such as would occur when maneuvering, would reduce $N'_{\delta_W}/L'_{\delta_W}$ to values less than 0.03. This fact, plus the repeated pilot commentary that yaw coupling due to roll <u>rate</u> [which is what $\delta'_r(3)$ is all about] was extremely objectionable should have made it obvious that the $\delta'_r(3)$ parameter should at least be checked. Since the publication of Ref. 6 we have increased the applicable range of $\delta'_r(3)$ from $|N'_{\delta_W}/L'_{\delta_W}| > 0.03$ to 0.07 with the provision that both μ and $\delta'_r(3)$ should be checked and the worst case utilized. It is hoped that this will remove the necessity to apply engineering judgment in the selection of the appropriate criterion.

A benefit from the Ref. 14 critique is that we now have a data point in a region where no data were previously available. It is gratifying that this new point supports the extrapolated boundary shown in Fig. 6.

There is continued implication throughout the Ref. 14 critique that a first-order model must be identified before the μ criterion may be applied. This is erroneous. As stated in this paper and as was stated in Ref. 6, high-frequency roots (at or above 6 rad/sec) which occur in pairs (denominator and numerator) should be removed from Y_{CF}. No other modification is required. A simplified crossfeed which also removes the low-frequency roots is used in Fig. 4 of Ref. 14 to calculate $\delta_r(3)$. This is nonstandard usage which if corrected would modify the (Ref. 14) Fig. 4 time history; accordingly, the Ref. 14 criticism that the simplified (Fig. 4, Ref. 14) crossfeed provides nearly perfect coordination only out to 7 sec would be modified somewhat. However, as stated in the μ development paper, abrupt aileron inputs longer than 3 sec are of no interest for closed-loop heading control. We would therefore consider the β time history shown in Fig. 4, Ref. 14 representative of excellent coordination for the precision heading control task.

The second example cited by Calspan (an early version of the YF-16) also turned out to be quite beneficial in terms of lending additional insight into application of the μ parameter. In this case the required rudder to coordinate was found to be extremely non-monotonic (Fig. 8, Ref. 14). The general nature of the shape of the actual required rudder and the shape suggested by the lower-order equivalent system defined by μ is shown in Fig. 7. The extreme mismatch between the HOS and LOS precludes even a cursory evaluation of μ. However, the complex nature of the required rudder to coordinate a step aileron input would in itself lead one to suspect very poor pilot opinion of heading control. While Calspan was not able to produce a pilot rating for this configuration, it is well known that the original version of the YF-16 was an extremely poor airplane (pilot ratings of 9 and 10). As mentioned in the paper, no data are available to date to allow us to extend the longitudinal results of Ref. 7 to the lateral-directional axis, e.g., that the existence of a very poor LOS match is in itself a measure of poor handling. These data, rather than invalidating μ, actually provide the first available data which tend to support the assumed extension of the Ref. 7 results.

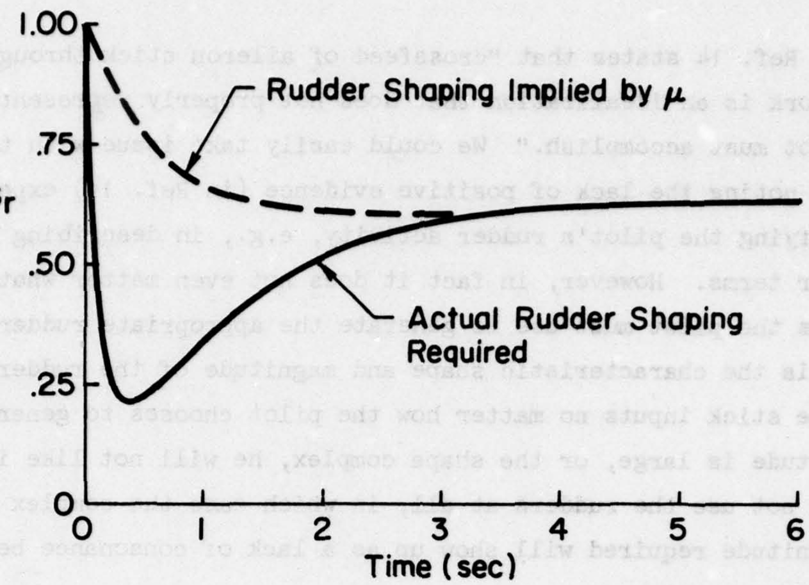

Figure 7. Rudder Time Histories

As noted by Calspan in their presentation, some of the ψ_β values used by STI in Ref. 6 were in error. This issue was covered in correspondence between Calspan and STI over a year ago. However, the tendency to miscalculate ψ_β is perhaps an inherent deficiency in the parameter. Witness the discrepancies for the same flight conditions between the values of ψ_β which appear in two separate Calspan authored reports:

Configuration	ψ_β (deg) Reference 4	Reference 11
2P2	−295	−254
3NO	−189	−224
3P2	−344	−290
4P2	−332	−208
12A2	−207	−159
12A1	−210	−167
12P2	−356	−291

No attempt has been made to determine which of these represents the "correct" values of ψ_β.

Finally, Ref. 14 states that "crossfeed of aileron stick through a shaping network is an idealization that does not properly represent the task the pilot must accomplish." We could easily take issue with this statement by noting the lack of positive evidence (in Ref. 14) experimentally quantifying the pilot's rudder activity, e.g., in describing function or other terms. However, in fact it does not even matter what internal feedbacks the pilot must use to generate the appropriate rudder; what does matter is the characteristic shape and magnitude of the rudder required to coordinate stick inputs no matter how the pilot chooses to generate it. If the magnitude is large, or the shape complex, he will not like it. In fact, he may not use the rudders at all, in which case the complex shaping or large magnitude required will show up as a lack of consonance between bank angle and yaw rate.

QUESTIONS AND ANSWERS

1. Ralph Smith: Is there a difference in short time tracking performance from adverse to proverse yaw at the corresponding Level 1 boundaries?

Yes. Short term tracking performance was best for $\mu = -1$ and proverse $N_{\delta_W}/L_{\delta_W}$. Physically the heading response leads the bank angle in this region which tends to give abrupt lateral motion at the cockpit but results in very tight tracking.

2. Chick Chalk: How is $\delta_r'(3)$ normalized when $\left|\frac{N_{\delta_W}}{L_{\delta_W}}\right| < 0.03$.

$\delta_r'(3) = N_{\delta_r}'\delta_r(3)$. Implication is that N_{δ_r}' is removed and can be separately optimized. Also note that $\delta_r'(3)$ is calculated any time $\left|\frac{N_{\delta_W}'}{L_{\delta_W}'}\right| < 0.07$.

3. Dwight Schaeffer, Boeing: What happens if a lot of β is needed to coordinate turn?

Coordinated turn is defined as when yaw and roll are in consonance, e.g., $r = (g \sin \varphi)/V$. We have assumed that this is well approximated when β and $\dot{\beta}$ are zero, viz.,

$$(s - Y_v)\beta + r + \frac{g \sin \varphi}{V} = Y_{\delta_r}\delta_r + Y_r r$$

Underlying assumption is $Y_{\delta_r} + Y_r \doteq 0$.

4. Jerry Lockenour, Northrop: It would be instructive to show δ_r coordination for aileron pulse instead of a step because β ($\delta_{r_{coord}}$) transient upon input removal is important. Also for depressed reticle bomb sights sideslip upon aileron removal can be helpful in minimizing the "pendulum effects." Any comment? Did you consider L_{δ_r}?

a. Our philosophy in showing an aileron step is that it is illustrative of the pilots rudder action required to initiate a bank. Once the bank is established, the removal of aileron is considered to be a step in the opposite direction.

b. The use of sideslip to quicken the heading response shows up in the μ parameter as a "bulge" allowing large proverse $N'_{\delta_w}/L'_{\delta_w}$ at $\mu = -1$.

c. L_{δ_r} is implicit in the β numerators. The answer is therefore, yes.

CALSPAN ADVANCED TECHNOLOGY CENTER

8 September 1978

PROPOSED REVISION TO THE LATERAL-DIRECTIONAL COUPLING REQUIREMENTS OF MIL-F-8785

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PROPOSED REVISION TO THE LATERAL-DIRECTIONAL COUPLING REQUIREMENTS OF MIL-F-8785

INTRODUCTION

The military flying qualities specification MIL-F-8785B(ASG) was adopted in 1968. In the intervening years the specification has been applied by designers, government procurement agencies and flight test organizations. The Air Force Flight Dynamics Laboratory has sponsored a number of contracts with airplane design organizations to make detail comparisons of existing aircraft F-4, C-5, F-5, and P-3 with MIL-F-8785B. Also a number of research contracts have been let to develop and propose revisions to the requirements. Under two of these contracts, attention was directed at the lateral-directional coupling requirements in MIL-8785 and revisions to those requirements have been proposed for consideration in References 1 and 2. The purpose of this paper is to present the proposed revisions developed by Calspan Corporation in Reference 1 and to compare the requirements for limiting sideslip proposed by Calspan with the aileron-rudder crossfeed requirement proposed by System Technology Incorporated in Reference 2.

DISCUSSION OF LATERAL-DIRECTIONAL REQUIREMENTS PROPOSED BY CALSPAN

The lateral-directional coupling requirements proposed by Calspan and the associated terminology definitions are included as Appendix A to this paper.

Bank Angle and Roll Rate Oscillatory Requirements

Requirements to limit bank angle and roll rate oscillations resulting from the pilot's use of aileron were introduced in MIL-F-8785B(ASG). These requirements have now been tested by additional data and have been subjected to the test of application. From this experience it is concluded that the requirements are basically valid and certainly should be retained, but certain

problems in the present statement of the requirements have come to light which should be rectified.

1. It has been found that certain combinations of stability derivatives result in $\measuredangle \frac{p}{\beta}$ of the Dutch roll mode between 180^{o} and 270^{o}. Because of this, the use of two ψ/β scales in the requirement and the rule for deciding which scale is to be used based on $\measuredangle \frac{p}{\beta}$ has been re-examined and found to be an inadequate feature of the requirement. Both this problem and the problem identified in Item 2 can be remedied by using ψ/p rather than ψ/β in the requirement.

2. When $\left|\phi/\beta\right|_{d}$ is large, the amplitude of the Dutch roll oscillation in sideslip is quite small and measurement of the phase angle is difficult and inaccurate.

3. When the spiral root is not at the origin, the measures ϕ_{osc}/ϕ_{AV} and p_{osc}/p_{AV} are severely distorted and do not correlate with the flying qualities rating. This problem can be remedied by removing the spiral mode contribution to the ϕ and p time histories.

4. The symmetrical shape of the requirement boundary is inaccurate; additional data permit defining the boundary more accurately.

5. The requirements on roll rate oscillations require step aileron inputs to be held for 1.7 T_d sec in one case and until bank angle has changed 90^{o} in another case. To avoid extreme roll attitude at the end of the test, it is desirable to start the maneuver from banked turning flight in one direction and to roll through wings level to bank in the opposite direction. The requirement that the rudder pedal be free during this maneuver prevents setting up zero-sideslip initial conditions or if the rudder is used to zero initial

sideslip and is then released when the aileron step is applied, the resulting response is not to the aileron input alone. This problem can be remedied by specifying that the rudder pedal be held fixed for the requirements that are based on aileron step inputs.

6. The large difference in the magnitude of p_{osc}/p_{AV} permitted by the requirement for $\psi_\beta = -240^o$ relative to that permitted for $\psi_\beta = -40^o$ is primarily a result of the fact that p_{AV} tends to zero for large adverse yaw due to aileron. This causes p_{osc}/p_{AV} to tend to infinity, but the pilot rating does not degrade nearly so rapidly, thus the gradient of pilot rating with p_{osc}/p_{AV} is shallow, making location of Level 2 boundaries imprecise. This situation can be alleviated by changing the denominator of the ratio to $\hat{p}_1$ and $\hat{\phi}_1$, respectively,in the definitions of $\hat{p}_{osc}/\hat{p}_1$ and $\hat{\phi}_{osc}/\hat{\phi}_1$.

7. The formula using two peaks to calculate p_{osc}/p_{AV} and ϕ_{osc}/ϕ_{AV} when $\zeta_d > .3$ has been found to have a characteristic which results in lack of discrimination for certain $\psi_{p_{step}}$ or $\psi_{\phi_{impulse}}$ phase angles. This situation can be avoided by using a formula based on three peaks for all cases regardless of the value of the Dutch roll damping ratio. When ζ_d is high and there is no third peak, $p_3 = p_2$.

Sideslip Excursion Limit Requirements

The roll oscillation requirements are quite effective in limiting Dutch roll excitation resulting from control of roll rate and bank angle when the roll coupling derivatives L'_r and L'_β are significant. When these derivatives are small, however, the roll oscillation requirements are ineffective in limiting the Dutch roll excitation which will be manifested as a yawing oscillation with little roll. The flying qualities problems for configurations

of this type are most serious if the Dutch roll frequency is low and the damping is light. For the situation of low roll-yaw coupling, it is important to limit the excitation of the Dutch roll mode in sideslip and yaw rate resulting from the pilot's use of aileron to control roll rate, bank angle and heading.

In MIL-F-8785B(ASG) the parameter $\Delta\beta_{max}/k$ versus ψ_{β} step was introduced to limit the Dutch roll excitation in sideslip resulting from a step aileron input. The requirement has several undesirable features.

1. The requirement is based on step aileron inputs up to the magnitude that causes 60 degree bank change in a time of 2 sec or T_d sec, whichever is longer. For low frequency Dutch roll roots, it is necessary to restrict the size of the aileron input to quite small values.

2. For some configurations, the maximum sideslip occurring within T_d sec results from the residue of the spiral mode rather than the Dutch roll mode. For landing approach data, the resulting sideslip does not correlate with pilot rating data.

3. The definition of the parameter $K = \frac{\phi_{t_{command}}}{\phi_{t_{requirement}}}$ involves the roll performance requirement. Thus, the validity of the sideslip requirement depends on the validity of the roll performance requirements.

4. Permitting use of rudder to determine $\phi_{t_{command}}$ in a requirement based on rudder-pedal-free aileron inputs leads to confusion and additional work. This is a cumbersome aspect of the requirement because it requires a series of tests or analyses to define the roll performance as a function of aileron input amplitude and a separate series of tests or analyses to determine the sideslip excursions as a function

of aileron input amplitude. For analytical checks of the requirement there is no guidance as to how to calculate $\phi_{t_{command}}$. This aspect of the requirement is not substantiated in the BIUG because none of the data presented was treated in that way. The data were reduced in the form $\Delta\beta_{max}/\phi_{t=1}$ for step aileron inputs and then multiplied by what was considered to be the appropriate $\phi_{t=1}$ requirement for each experiment with no consideration being given to use of rudder to prevent adverse yaw.

5. Because the ϕ_t required values are different for Levels 1 and 2, it is necessary to calculate two separate values of $\Delta\beta_{max}/k$ for each data point to be compared with the requirements, one for Level 1 which is compared with the Level 1 boundary and one for Level 2 which is compared with the Level 2 boundary. This "double relaxation" of the requirement, i.e., lower required roll performance and a separate Level 2 boundary in the $\Delta\beta_{max}/k$ versus ψ_β plane, is involved and cumbersome and is not shown to be necessary by the available data.

6. Although the $\Delta\beta_{max}/k$ requirement in Figure 6 of MIL-F-8785B(ASG) has three boundaries, one for Level 2 and two for Level 1 as a function of Flight Phase Category, the requirement is actually the most complex and finely defined requirement in the entire specification. This is because the parameter k depends on the roll performance requirements which are specified as a function of Class, Flight Phase and Level. In the BIUG the parameter k is discussed as being the ratio of roll performance attainable from a specific aileron input to roll performance required by the specification and is presented as a replacement for the scaling rule in MIL-F-8785, paragraph 3.4.9. In paragraph 3.4.9, the sideslip resulting from a given input was scaled by the ratio of aileron deflection used to

the aileron deflection required to meet roll performance requirements.

If the $\Delta\beta_{max}/k$ parameter is written as $\phi_{t_{required}} \dfrac{\Delta\beta_{max}}{\phi_{t_{command}}}$ it seems that a better interpretation is that the ratio $\Delta\beta_{max}/\phi_{t_{command}}$ must be multiplied by a weighting constant $\phi_{t_{required}}$ which makes the requirement a function of Class, Flight Phase, and Level. In other words, the requirement could have been written such that the measurement $\dfrac{\Delta\beta_{max}}{\phi_{t_{command}}}$ is compared with a new Figure 6 which would have about 45 different curves on it, one for each $\phi_{t_{required}}$ value in 3.3.4 and 3.3.4.1. Some of the 45 curves would be coincident with Class I, Flight Phase Category A, Level 2 would be coincident with Class I, Flight Phase Category B, Level 1 because $\phi_{t_{required}}$ is 60^{o} in 1.7 sec for both situations. If the sideslip excursion requirement had been presented in the latter format, it would no doubt have received much more criticism.

The $\dfrac{\Delta\beta_{max}}{k}$ requirement of MIL-F-8785B(ASG) is tied to the bank angle response through a number of assumptions implicit in this requirement. The requirement assumes that a measure of the undesired response (sideslip excursion) resulting from an aileron input ratioed to the desired response (bank angle at a specific time) resulting from the aileron input will reflect flying qualities. The use of bank angle response as the measure of the desired response to aileron was intuitive, but to an extent arbitrary. In the roll oscillation requirements, the oscillatory component of roll rate or bank angle is ratioed to the average roll rate or bank angle resulting from the aileron input. This is a straightforward requirement relating an undesirable component of a response to the desirable component of the same response. In the case of the sideslip excursion requirement, the Dutch roll component of sideslip is identified as an undesirable component of the airplane's response

to aileron commands but it is clearly an assumption to expect a measure of bank angle to be a universally valid parameter to use as the measure of the desired response to aileron. The situation is one of dividing "apples and oranges" so if a successful correlation parameter of this form does exist, its discovery will depend on intuition and empirical correlation. Using intuition one might argue that sideslip is a yawing motion and should therefore be ratioed with an aspect of the airplane heading or yaw rate response to aileron. Based on this intuition and examination of $\Delta\beta/\phi$ type measures from several experiments, it was determined that better correlation of several sets of data could be obtained by introducing the factor $\frac{g}{V_T}$. Specifically, the following parameter has been found to have potential as a flying qualities parameter.

For aileron impulse:

$$\frac{\left|\Delta\beta_{max}\right|_{t < 1.2\ T_d}}{\omega_d \left[\frac{g}{V_T}\ \hat{\phi}_1\right]} \quad \text{versus} \quad \psi_{\beta\,\text{impulse}}$$

The Laplace transform for a step command is 1/s and the transform for an impulse command is unity. As a result the inverse transform of a variable to an impulse input is identical to the inverse transform of the derivative of the variable for a step input. Therefore,

For aileron step:

$$\frac{\left|\Delta\dot{\beta}_{max}\right|_{t < 1.2\ T_d}}{\omega_d \left[\frac{g}{V_T}\ \hat{p}_1\right]} \quad \text{versus} \quad \psi_{\dot{\beta}\,\text{step}}$$

Advantage is also taken of this fact in stating the requirements for bank angle oscillations for an impulse input and the requirements for roll rate oscillations for a step input. In this example

$$\left.\frac{\hat{p}_{osc}}{\hat{p}_1}\right|_{step} = \left.\frac{\hat{\phi}_{osc}}{\hat{\phi}_1}\right|_{impulse} \quad \text{and} \quad \psi_{p_{step}} = \psi_{\phi_{impulse}}$$

For an impulse aileron input, the sideslip response will consist mainly of the Dutch roll component with no problems from the spiral mode unless it is highly divergent. Thus, the measure of sideslip is $\Delta\beta_{max}$ defined as the maximum peak-to-peak excursion occurring within 1.2 T_d seconds. This time interval is stated to eliminate problems with an unstable Dutch roll mode or a highly unstable spiral mode. This measure of sideslip is divided by $\frac{g}{V_T}\hat{\phi}_1$, which can be viewed as a measure of the yaw rate response to the aileron input. $\hat{\phi}_1$ is the first peak of the bank angle response. A measure like $\hat{\phi}_{AV}$ could be specified to better reflect the yaw rate resulting from the steady bank angle but this was not considered likely to be significantly different from $\hat{\phi}_1$ for the low $|\phi/\beta|_d$ cases for which the sideslip excursion requirements are critical. The parameter $\hat{\phi}_1$ is proposed to eliminate the possibility that there would not be a peak if the spiral mode were unstable.

The residue of the Dutch roll mode in the sideslip response to a step aileron input is roughly proportional to $\omega_{n_d}^{-2}$, however, the residue for an impulse input is roughly proportional to $\omega_{n_d}^{-1}$. Because of this characteristic, it is necessary to divide $\Delta\beta_{max}$ measured from an impulse input by ω_{n_d} in order to preserve the empirically demonstrated correlation between pilot rating and the $\Delta\beta_{max}/k$ criterion in MIL-F-8785B(ASG). To clarify this point, assume that pilot rating is found by experiment to be a function of $\Delta\beta_{osc}$ for a step aileron input, P.R. = $f(\beta_{osc})_{step}$. Then it follows that pilot rating should be correlated with $\Delta\beta_{osc} \times \omega_{n_d}^{-1}$ when an impulse aileron input is used, P.R. = $f\left(\frac{\Delta\beta_{osc}}{\omega_{n_d}}\right)$ impulse. This is the rationale for including $\omega_{n_d}^{-1}$ in the definition of the new sideslip parameter proposed by Calspan.

COMPARISON OF CALSPAN SIDESLIP REQUIREMENT AND STI AILERON-RUDDER CROSSFEED PARAMETERS

The $\Delta\beta_{max}/k$ vs $\psi_{\beta_{step}}$ requirement of MIL-F-8785B(ASG) was criticized by the authors of References 2 and 3 as being "....based on aileron-only parameters and the effects of rudder control are only indirectly apparent as they may have influenced individual pilot ratings." The authors of

References 2 and 3 further claimed, "the fact that these criteria are not satisfactory is shown in Figure 10 where several configurations which violated boundaries based on aileron-only parameters were given good to excellent pilot ratings." The data and figure referred to are reproduced herein as Figure 1. Specifically, data points 4B, 5A and 5B from Reference 4 were singled out as gross violations of the $\Delta\beta_{max}/k$ requirement and held up as proof that the $\Delta\beta_{max}/k$ requirement was inadequate to handle cases which required the pilot to use the rudder for coordination. In fact, however, the violations of data points 4B, 5A and 5B illustrated in Figure 1 are fictitious because these points, and others in Figure 1, have been plotted by the authors of Reference 2 at the wrong values of ψ/β_{step}. The errors in locating these points relative to the specification boundaries are indicated on Figure 1. It is seen that when the points are properly plotted, they are not in gross violation but are quite well accommodated by the specifications requirement boundary. Although there are many reasons for making revisions to the $\Delta\beta_{max}/k$ requirement which were discussed elsewhere in this paper, the claim made by the authors of References 2 and 3 that the requirement does not account for the pilot's use of rudder to coordinate turns is quite unfounded. The evaluation pilots were free to use the rudder during all of the experiments and their ratings reflect the feasibility of using rudder to coordinate maneuvers. The shape of the requirement boundary as a function of ψ/β_{step} is partly determined by this consideration.

Calspan in Reference 1 and STI in Reference 2 have taken two different approaches to development of requirements for limiting sideslip during rolling and turning maneuvers. The time histories in Figure 2 illustrate two aspects of the problem and suggest the two approaches. Figure 2 shows the bank angle and sideslip responses for a given configuration that result from a step aileron input with rudder zero. Also shown on Figure 2 is the rudder input that must accompany an aileron step input in order to maintain zero sideslip. The bank angle response to the combination aileron and rudder input is identical to that for the aileron step alone for this configuration because the roll yaw coupling is very low. The Dutch roll mode is excited by

the aileron-alone input and has a residue in the sideslip, yaw rate and heading responses. The Calspan approach, described previously, is to develop requirements to limit the magnitude of the oscillatory component of the sideslip response to an impulse aileron input as a function of the phase angle of the oscillation. The STI criterion is based on the aileron-rudder sequencing required to achieve coordinated turns, defined as turns with zero sideslip. The ideal aileron-rudder crossfeed is the ratio of transfer function numerators $Y_{CF} = -N^{\beta}_{\delta_{AS}} / N^{\beta}_{\delta_{RP}}$.

The STI criterion is based on the assumption that the ideal rudder crossfeed transfer function can be adequately represented by a first order lead-lag form $Y_{CF} \cong -\frac{K(s + \lambda_z)}{(s + \lambda_p)}$. Under this assumption (which will later be shown to be invalid for the YF-16) a parameter, μ, is defined as the ratio of the separation of the zero from the pole normalized by the value of the pole of the lead-lag network. The STI criterion consists of empirical boundaries drawn on a plane defined by the μ parameter and the ratio of $N'_{\delta_a} / L'_{\delta_a}$ calculated for stability axes. Under the assumption that the crossfeed transfer function can be approximated by a lead-lag, the μ parameter can be defined in terms of the initial and final values of a time history of rudder pedal for a step aileron stick input to the simplified lead-lag transfer function, $\mu = \frac{\text{final value}}{\text{initial value}} - 1$.

In general the ideal aileron-rudder crossfeed transfer function will not be a first order lead-lag. For an unaugmented airplane the transfer function can be the ratio of third order polynomials and for the augmented YF-16 airplane the crossfeed is the ratio of sixth order polynomials. The authors of Reference 2 define a set of rules for simplifying higher order crossfeed transfer functions to the first order form. These simplification rules will be illustrated and the STI requirement will be compared with the Calspan requirement through application to two examples.

The first example is configuration P-8 taken from Reference 5. The ideal crossfeed transfer function for this example, together with a time

history for a step aileron wheel input, is shown on Figure 3. Following the rules for simplifying this transfer function given in Reference 2, the simplified transfer function and time history shown on Figure 3 is obtained. Evaluation of the μ parameter from the simplified transfer functions and time history is also shown on Figure 3.

The effect of the simplified crossfeed on the sideslip response of configuration P-8 is illustrated in Figure 4. In this case, the simplified crossfeed is effective in eliminating excitation of the Dutch roll mode and in constraining the sideslip to be nearly zero for the first seven seconds. After seven seconds the effect of eliminating the low frequency factors of the crossfeed is seen to result in an increase in the sideslip.

The roll rate and sideslip rate responses of configuration P-8 to a step aileron stick input without the crossfeed are shown in Figures 5 and 6. In Figure 5 it is seen that the p response exhibits a significant residue of the unstable spiral mode and measurement of p_{OS}/p_{AV} of MIL-F-8785B(ASG) would be severely influenced by the spiral mode. The $\hat{p}$ time history has been calculated by subtracting the contribution of the spiral mode as was described earlier. The reduced time history has been analyzed to calculate the parameters

$$\frac{\hat{p}_{os}}{\hat{p}_1} \text{ vs } \psi_{p_{step}}$$

recommended by Calspan in Reference 1. The calculations are illustrated in Figure 5. In Figure 6 it is seen that the $\dot{\beta}$ response to the step aileron stick input is dominated by the excitation of the Dutch roll mode whereas the β response to the step input illustrated in Figure 4 was highly influenced by the spiral mode residue. The calculations to evaluate the new sideslip coupling requirement recommended by Calspan are illustrated on Figure 6.

Now that the Calspan and STI recommended flying qualities parameters have been calculated for configuration P-8, they can be compared to the criteria boundaries in Figures A4, A5 and 7. Configuration P-8 was evaluated

by three different evaluation pilots and was rated as follows: Pilot A PR = 8, Pilot B PR = 9-10, Pilot C PR = 6-6.5. Unfortunately,none of the requirements are successful in screening out this configuration. It should be pointed out that a configuration that is rated PR > 6.5 does not have to appear outside the 6.5 boundary on all the flying qualities parameter plots; it only need exceed that boundary on one plot to have been properly screened out. In this case, the pilot complaints were directed at the sideslip and heading control difficulties and the large and difficult-to-accomplish rudder inputs required to prevent these problems. Therefore, one should not expect the $\hat{p}_{os}/\hat{p}_1$ requirement to screen out this case but it should be screened out by the Calspan sideslip requirement or the STI aileron-rudder crossfeed requirement. Actually this case was chosen by the author because it was anomalous in the Calspan data correlations reported in Reference 1 and was considered to be a test case of interest in comparing the STI and Calspan requirements. Extensive data correlations treating the data from fifteen flying qualities experiments are shown in Reference 1 but are too voluminous to report here.

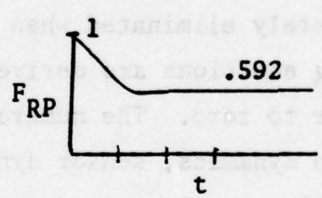
The second example chosen to compare the Calspan and STI requirements is the YF-16 as it was known to Calspan at the time Calspan simulated the aircraft in the T-33 VSS airplane prior to its first flight, Reference 6. This aircraft is augmented in a manner which results in higher order transfer functions and for this reason it is an interesting case to use in exploring the applicability of the proposed flying qualities requirements. The ideal aileron-rudder crossfeed transfer function for the YF-16 is the ratio of two sixth order polynomials in S. The rudder pedal force required for a step aileron stick force is shown in Figure 8. This is a complex time history involving a rapid rudder pedal force reversal within the first tenth of a second followed by a dip at t $\approx$ .4 sec and then a fairly steady growth with time. The complete crossfeed transfer function is noted on Figure 8. Using the STI rules for simplifying that transfer function, the reduced transfer function and time history illustrated on Figure 9 is obtained and μ = -.408. It should be recalled that the parameter was defined under the assumption that the crossfeed transfer function could be approximated by a first order lead-lag. Obviously the simplified YF-16 crossfeed which is third order in numerator

and denominator does not meet that assumption. Also, because of the second order complex factors in the numerator which cause a notch effect, the shape of the YF-16 crossfeed is quite different than that implied by a $\mu = -.408$ value for the first order lead-lag model. The first order model would imply a rudder pedal time history that starts at unity initial value and decays exponentially to the steady state value of .592. The time constant of the exponential decay is normally the roll mode time constant but in the case of the augmented YF-16, it is not obvious which of the three factors in the simplified crossfeed transfer function should be considered to be the roll mode. To make an extreme interpretation, the author assumed that the root at s + 1 should be used to relate the $\mu = -.408$ value to a first order crossfeed model. Then

$$\mu = \frac{\lambda_z}{\lambda_p} - 1 = \frac{.592}{1} - 1$$

The first order crossfeed model implied by $\mu = -.408$ then would be

$$Y_{CF} = \frac{.262\ (s + .592)}{(s + 1)}$$



where the gain has been calculated by rules given in Reference 3. This crossfeed was used to calculate the sideslip rate response illustrated in Figure 10. Obviously this crossfeed does not constrain sideslip to be zero and, in fact, by comparison with the sideslip rate time history in Figure 10 which was calculated for an aileron stick force step input with no crossfeed, it is seen that the first order crossfeed for which $\mu = -.408$ has aggravated the sideslip response. It is the author's opinion that this case demonstrates that the assumption that the crossfeed can be adequately represented by a first order lead-lag in the frequency band $.33 < w < 6$ is not valid and can lead to meaningless values of the μ parameter. This, of course, does not invalidate the aileron-rudder crossfeed approach to analyzing and evaluating flying qualities; it simply means that the first order model and the μ parameter are an oversimplification.

Although the YF-16 is highly augmented and the lateral-directional transfer functions are higher order, the p and $\dot{\beta}$ time histories in Figure 11 are amenable to evaluation of the flying qualities parameters recommended by Calspan in Reference 1. The calculations are illustrated on Figure 11. The parameters evaluated fall within the Level 1 requirement boundaries in Figures A4 and A5.

GENERAL OBSERVATIONS

It should be observed that the Calspan sideslip requirement and the STI crossfeed requirement are both related to the numerator of the β/δ_{AS} transfer function. Both requirements are aimed toward achieving airplanes that can be maneuvered in rolling and turning flight without sideslip. The information required to accomplish that objective is contained in the coefficients of the β/δ_{AS} transfer function numerator. The Dutch roll component of the sideslip response is minimized when the factors of the numerator cancel the Dutch roll roots of the characteristic equation. The sideslip response is completely eliminated when each of the numerator coefficients is zero. The following equations are derived by setting each coefficient of s in the β/δ_{AS} numerator to zero. The numerator polynomial being considered does not account for servo dynamics, sensor dynamics or electronic shaping networks. The equation applies to linearized equations of motion with augmented stability derivatives, including the artificial derivatives N_ϕ, L_ϕ and Y_ϕ which would result from feedback of bank angle to the rudder and aileron $N^{\beta}_{\delta_{AS}} = C_3S^3 + C_2S^2 + C_1S^1 + C_0$.

$$C_3 = 0 \text{ when } Y_{\delta_{AS}} = 0$$

$$C_2 = 0 \text{ when } N'_{\delta_{AS}} \cong \alpha_o L'_{\delta_{AS}} \qquad (1)$$

$$C_1 = 0 \text{ when } N'_p \cong \frac{g}{V} + \alpha_o (L'_p - N'_r) + Y_{\delta_{AS}} \frac{L'_p N'_r}{L'_{\delta_{AS}}}$$

$$C_0 = 0 \text{ when } N'_\phi = - \frac{g}{V} N'_r$$

These equations indicate that independent control of side force and yawing moments is required to eliminate the sideslip response to aileron stick. Side force control is necessary to cancel the side force due to aileron stick commands. This component is not normally a significant consideration and, for practical augmentation design, can be ignored. The remaining coefficients, C_2, C_1, C_0 can all be set to zero through generation of yawing moments with the rudder that are proportional to δ_{AS}, p and $\emptyset$. Expressed in equation form, the rudder augmentation required to constrain sideslip to be zero is:

$$\delta_{r_c} = \frac{\delta_{r_c}}{\delta_{AS}} \delta_{AS} + \frac{\delta_{r_c}}{p} p + \frac{\delta_{r_c}}{\emptyset} \emptyset \qquad (2)$$

The augmentation gains in this equation can be calculated from Equation 1. An example will be used to illustrate that an equation of this form will accurately represent the rudder time history calculated from the ideal aileron-rudder crossfeed transfer function. This process is illustrated in Figure 12. Note that it is necessary to have time histories for the p and $\emptyset$ (or at least $\emptyset$) responses to the aileron stick step which are calculated with the ideal aileron-rudder crossfeed included.

This interpretation of the rudder augmentation required to constrain sideslip to be zero suggests that crossfeed of aileron stick through a shaping network is an idealization that does not properly represent the task the pilot must accomplish. If he could fly the airplane in smooth air using only aileron stick inputs, then crossfeed of the stick inputs through the crossfeed filter to the rudder would prevent excitation of sideslip. The pilot, however, must fly the airplane in rough air and he may occasionally "miscoordinate" with the rudder. In these circumstances, the airplane motions are the result of more inputs than just the pilot's aileron stick commands. In this situation, the pilot must resort to the control law of Equation 2 which requires independent observation of δ_{AS}, p and $\emptyset$. In many cases that task is too demanding and the pilot may decide the best way to fly the airplane is not to attempt to use the rudder.

CONCLUSIONS

1. Although the lateral-directional coupling requirements of MIL-F-8785B(ASG) are generally valid, application experience and additional data has revealed deficiencies. These deficiencies have been addressed by Calspan in Reference 1 and revisions to the definitions of the parameters involved have been proposed which are designed to alleviate the observed deficiencies.

2. The sideslip requirement recommended by Calspan and the aileron-rudder crossfeed analysis proposed by STI are both useful in understanding the cause of flying qualities deficiencies. The μ parameter used in the STI heading control criterion, however, is based on the assumption that the aileron-rudder crossfeed transfer function can be adequately represented by a first order lead-lag network. This assumption is not generally valid and in some cases the magnitude of the μ parameter resulting from the assumption implies quite different rudder coordination from that actually required to restrain sideslip to be zero.

3. It is shown that the following rudder control law will constrain sideslip to be zero during rolling and turning maneuvers.

$$\delta_{r_c} = \frac{\delta_{r_c}}{\delta_{AS}} \delta_{AS} + \frac{\delta_{r_c}}{p} p + \frac{\delta_{r_c}}{\phi} \phi$$

The gains in this control law can be evaluated by setting the coefficients of $N^{\beta}_{\delta_{AS}}$ to zero.

4. Both the Calspan and STI requirements can be applied during the design phase of an aircraft development. The STI crossfeed criterion, however, would be difficult to evaluate

during the flight test phase. Evaluation would require identification of a complete set of stability derivatives from flight test data and use of these derivatives to show compliance through calculations. The Calspan requirements can be evaluated directly from recorded time histories of responses to step or impulse control inputs applied by the pilot.

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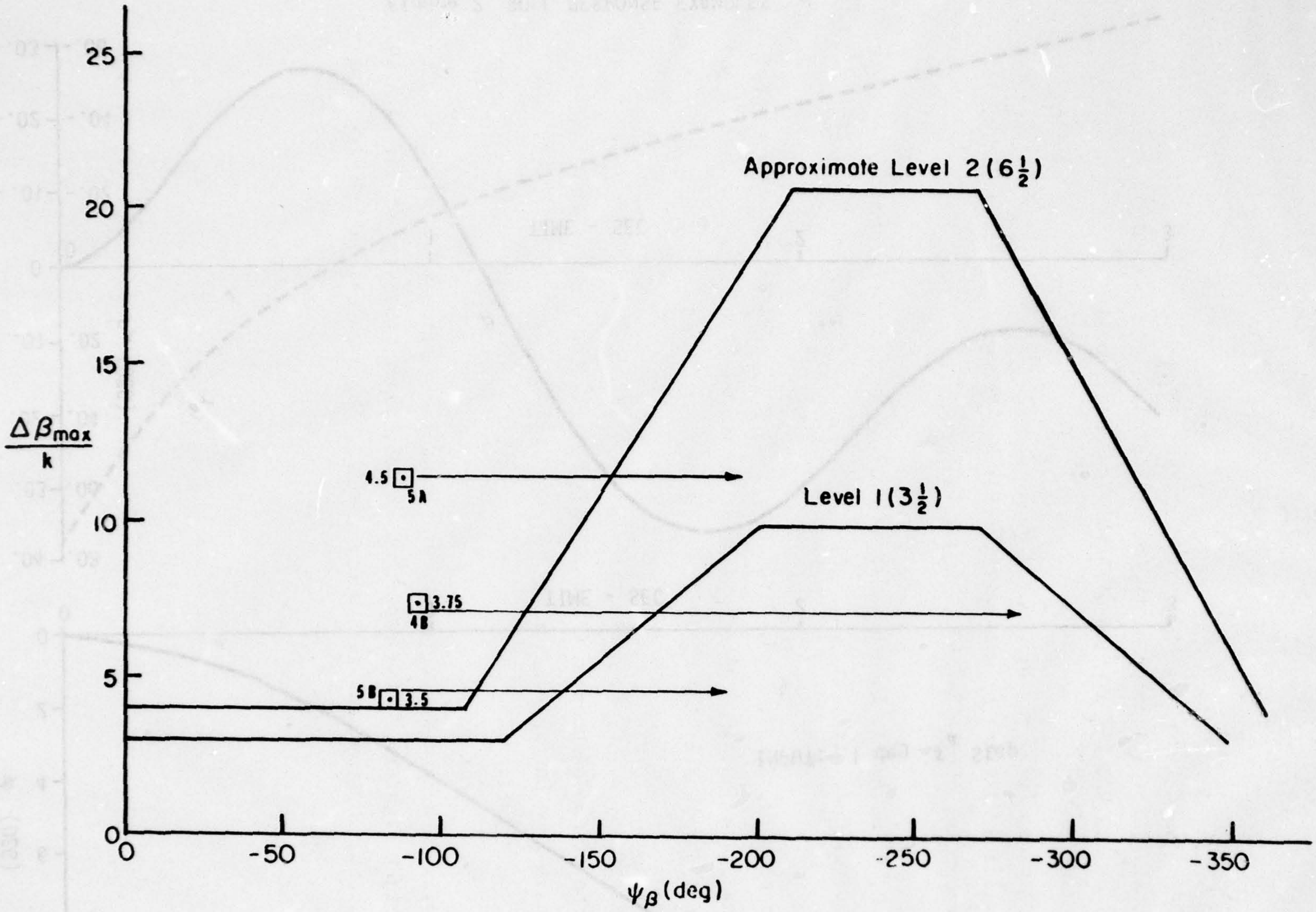


Figure 1 CORRELATION DATA ON $\Delta\beta_{MAX}/k$ vs ψ_β PLOT

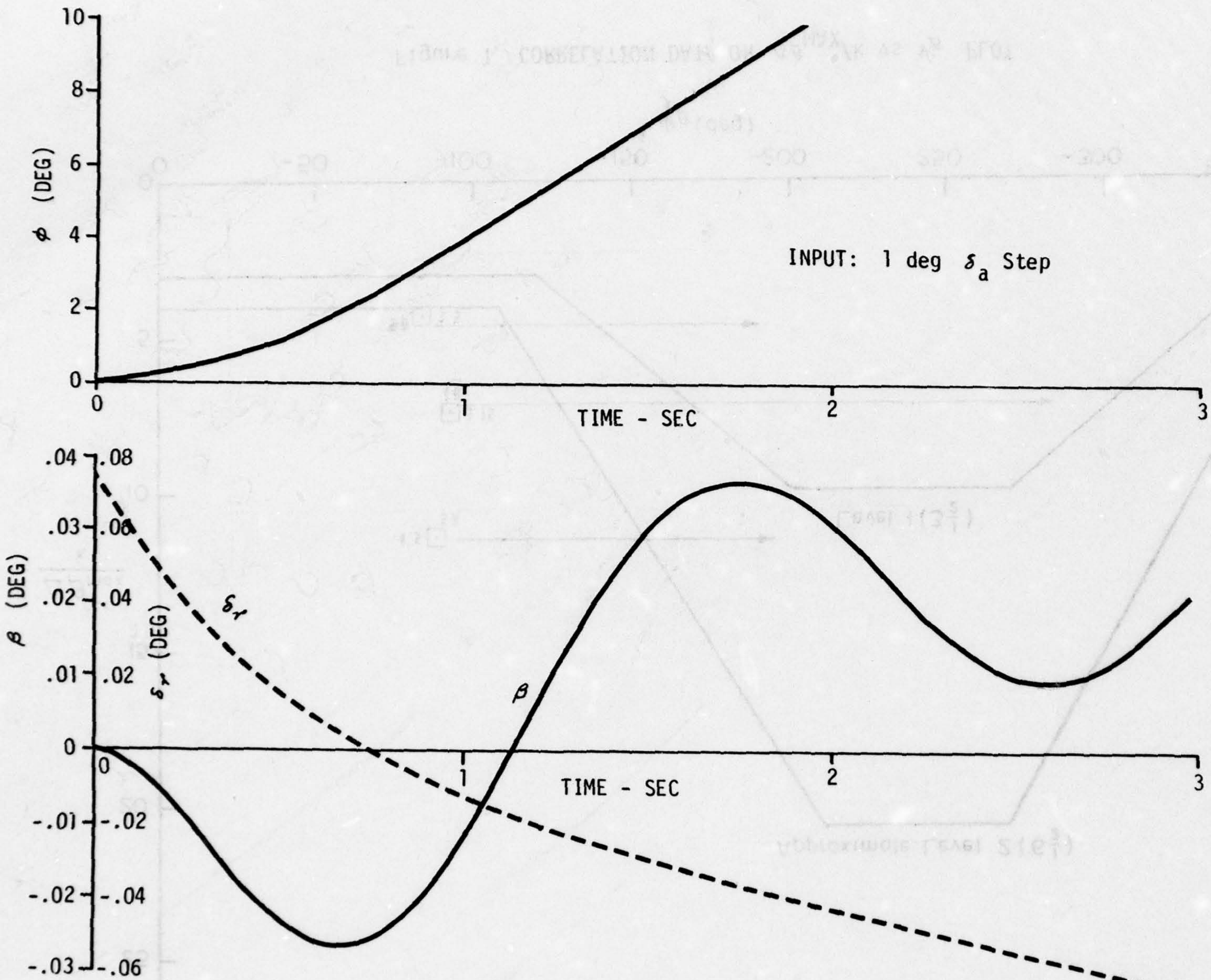


Figure 2 ROLL RESPONSE EXAMPLES

$$\text{- - -}\ \delta_{RP} = \frac{.177(S+.0799)(S-5.80)(S+23.59)}{(S-.0264)(S+3.40)(S+11.66)} \cdot \frac{\delta_W}{S}$$

$$\text{——}\ \delta_{RP} = \frac{(S-5.80)}{(S+3.40)} \cdot \frac{\delta_W}{S}$$

$$\mu = \delta_r(3) - 1$$
$$= -1.706 - 1$$
$$= -2.706$$

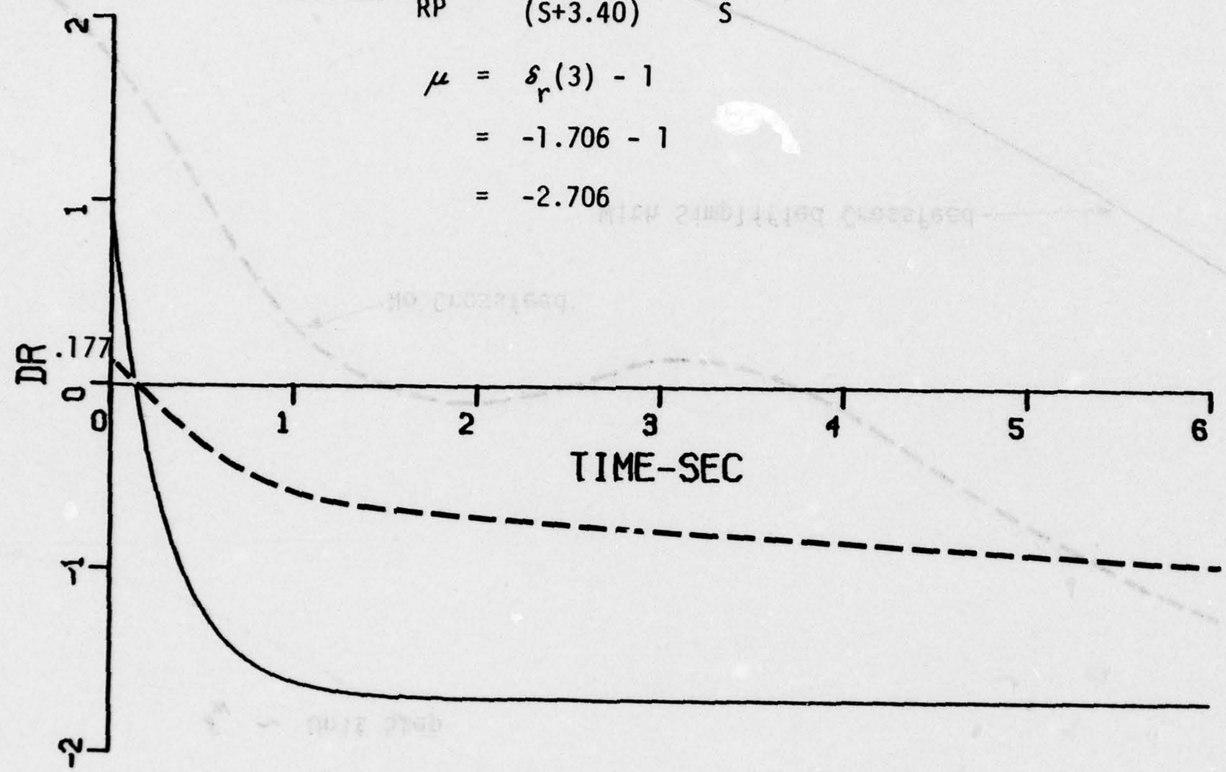


Figure 3 RUDDER PEDAL CROSSFEED FOR CONFIGURATION P-8

$$\beta(S) = \left[\frac{\beta}{\delta_W} + \frac{\beta}{\delta_{RP}} Y_{CF} \right] \delta_W (S)$$

$$Y_{CF} = \frac{.357(S-5.8)}{(S+3.4)} \quad \text{Simplified} \sim \text{STI rules}$$

$$\delta_W \sim \text{Unit Step}$$

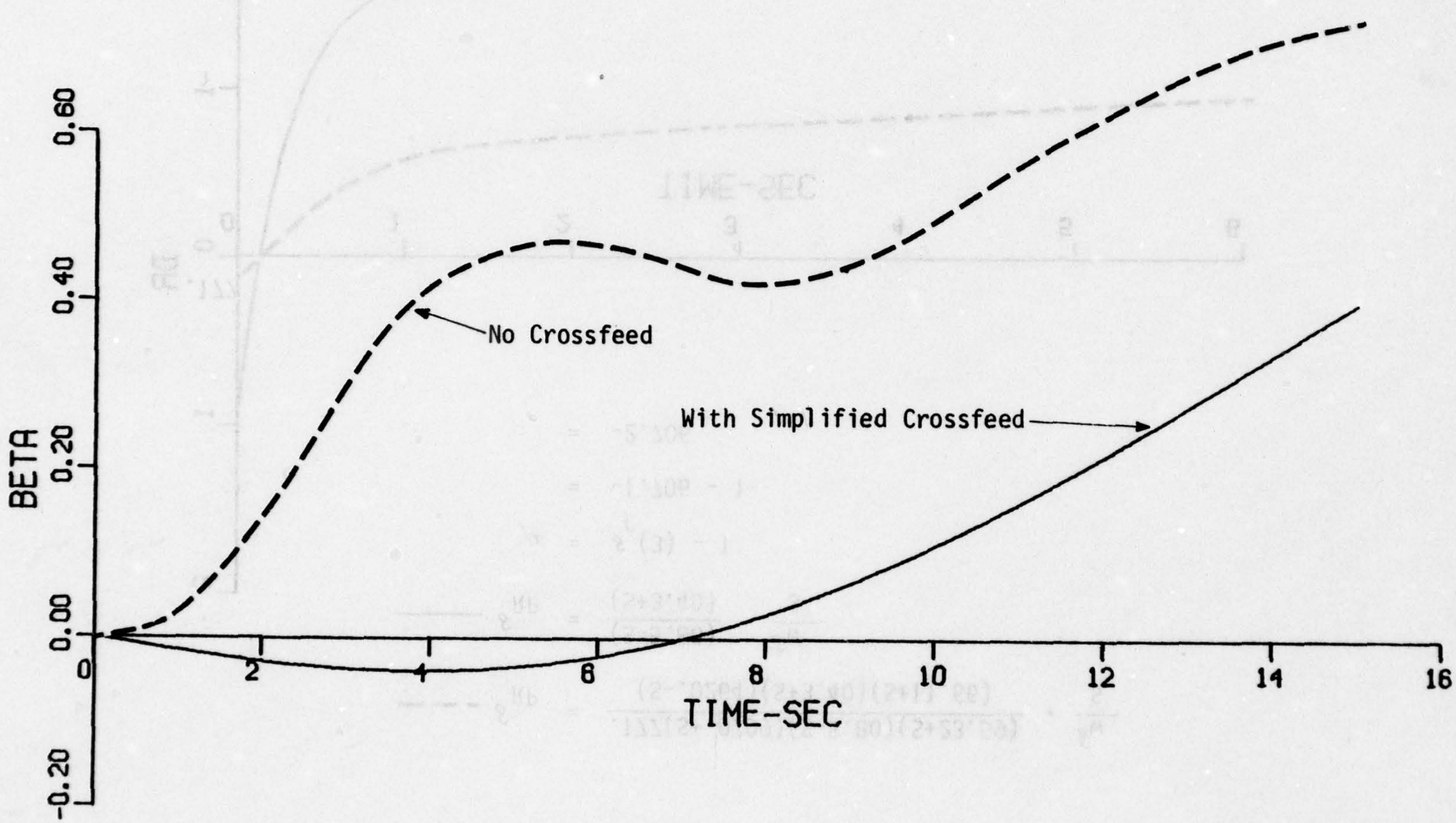
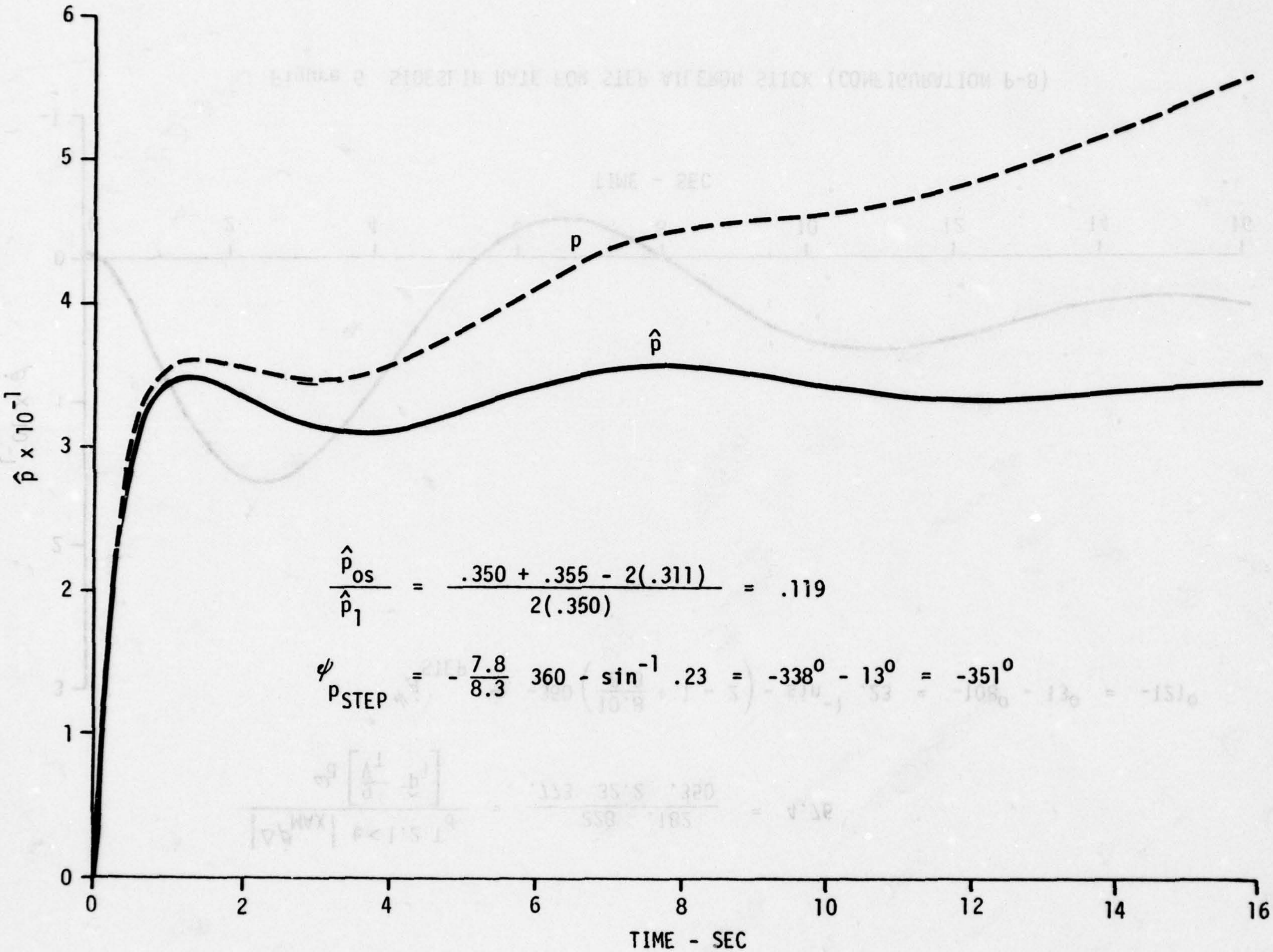


Figure 4 SIDESLIP RESPONSE OF CONFIGURATION P-8



$$\frac{\hat{p}_{os}}{\hat{p}_1} = \frac{.350 + .355 - 2(.311)}{2(.350)} = .119$$

$$\psi_{p_{STEP}} = -\frac{7.8}{8.3}\ 360 - \sin^{-1} .23 = -338^{o} - 13^{o} = -351^{o}$$

Figure 5 ROLL RATE FOR STEP AILERON STICK (CONFIGURATION P-8)

$$\frac{\left|\Delta\dot{\beta}_{MAX}\right| \; t < 1.2\;T_d}{\omega_d\left[\frac{g}{V_T}\;\;\hat{p}_1\right]} = \frac{228 \quad .182}{.773 \quad 32.2 \quad .350} = 4.76$$

$$\psi_{\dot{\beta}_{STEP}} = -360\left(\frac{10.8}{8.3} + 1 - 2\right) - \sin^{-1} .23 = -108^{o} - 13^{o} = -121^{o}$$

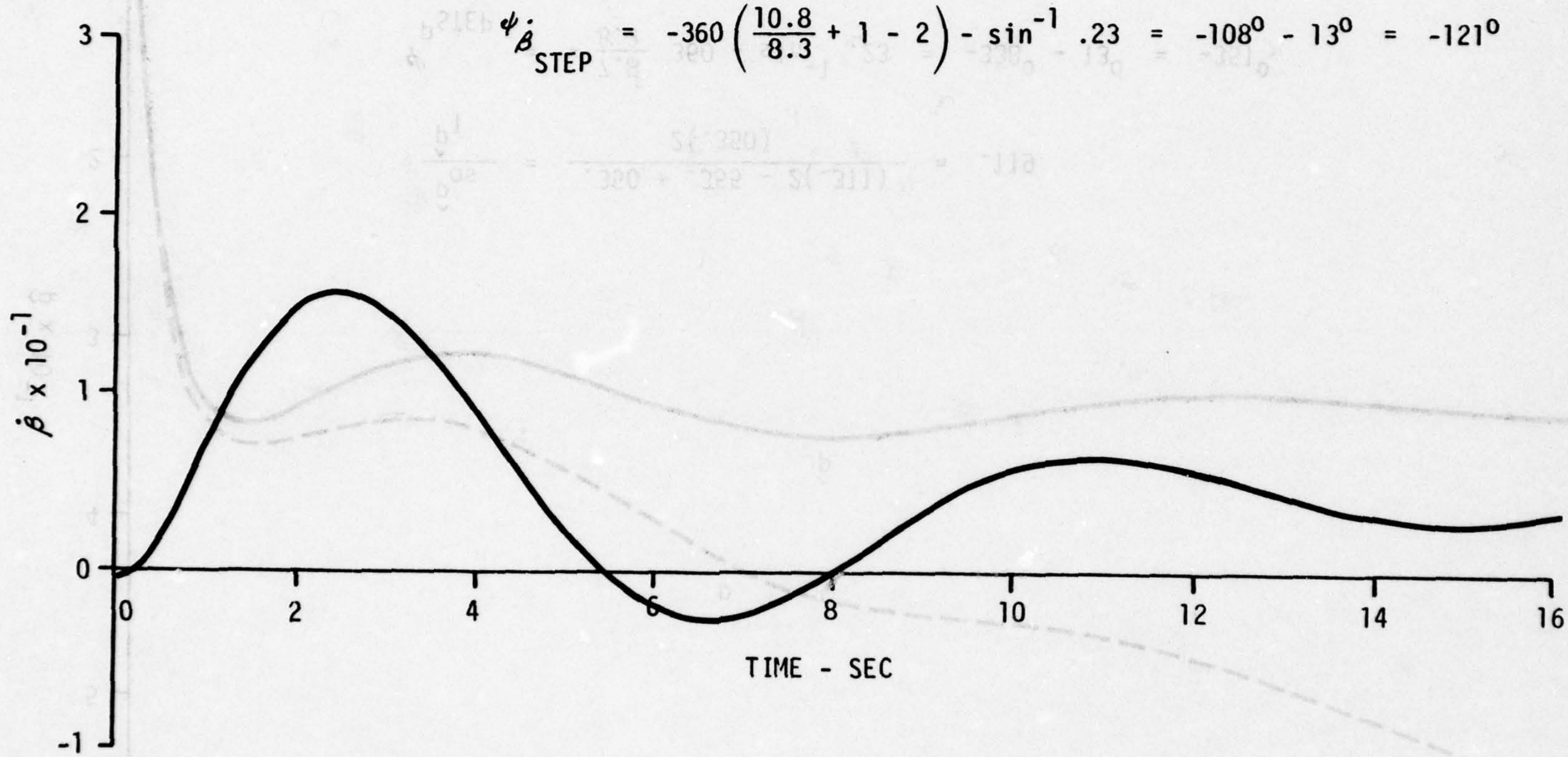


Figure 6 SIDESLIP RATE FOR STEP AILERON STICK (CONFIGURATION P-8)

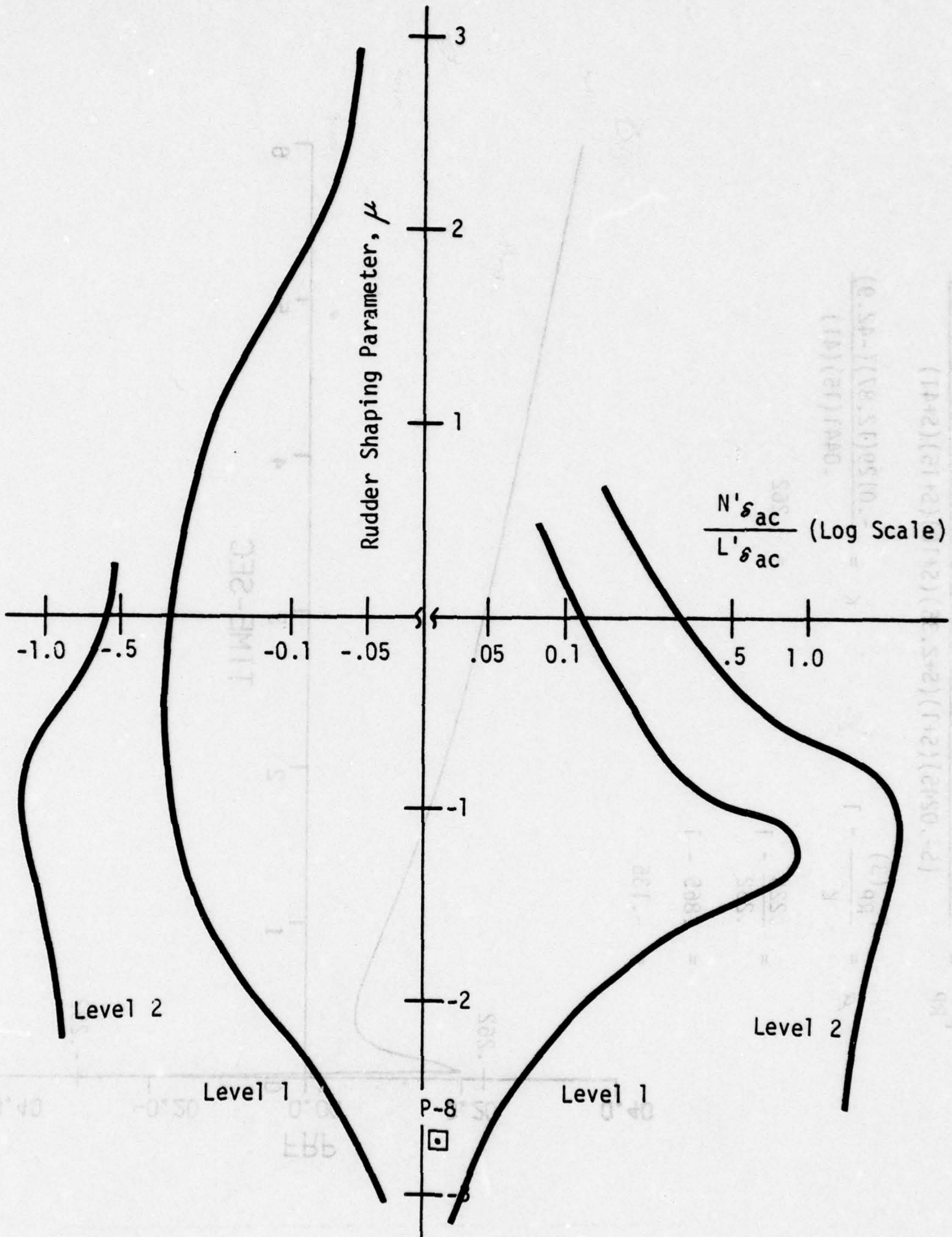


Figure 7 AILERON-RUDDER COORDINATION LIMITS

$$F_{RP} = \frac{-.292(S+.178)\left[S^2+.995S+3.748\right](S+4)(S+12.87)(S-42.9)}{(S-.0245)(S+1)(S+2.35)(S+10)(S+15)(S+41)}$$

$$\mu = \frac{F_{RP}(3)}{K} - 1 \qquad K = \frac{-.0129(12.87)(-42.9)}{.0441(15)(41)}$$

$$= \frac{.226}{.262} - 1 \qquad = .262$$

$$= .865 - 1$$

$$= -.135$$

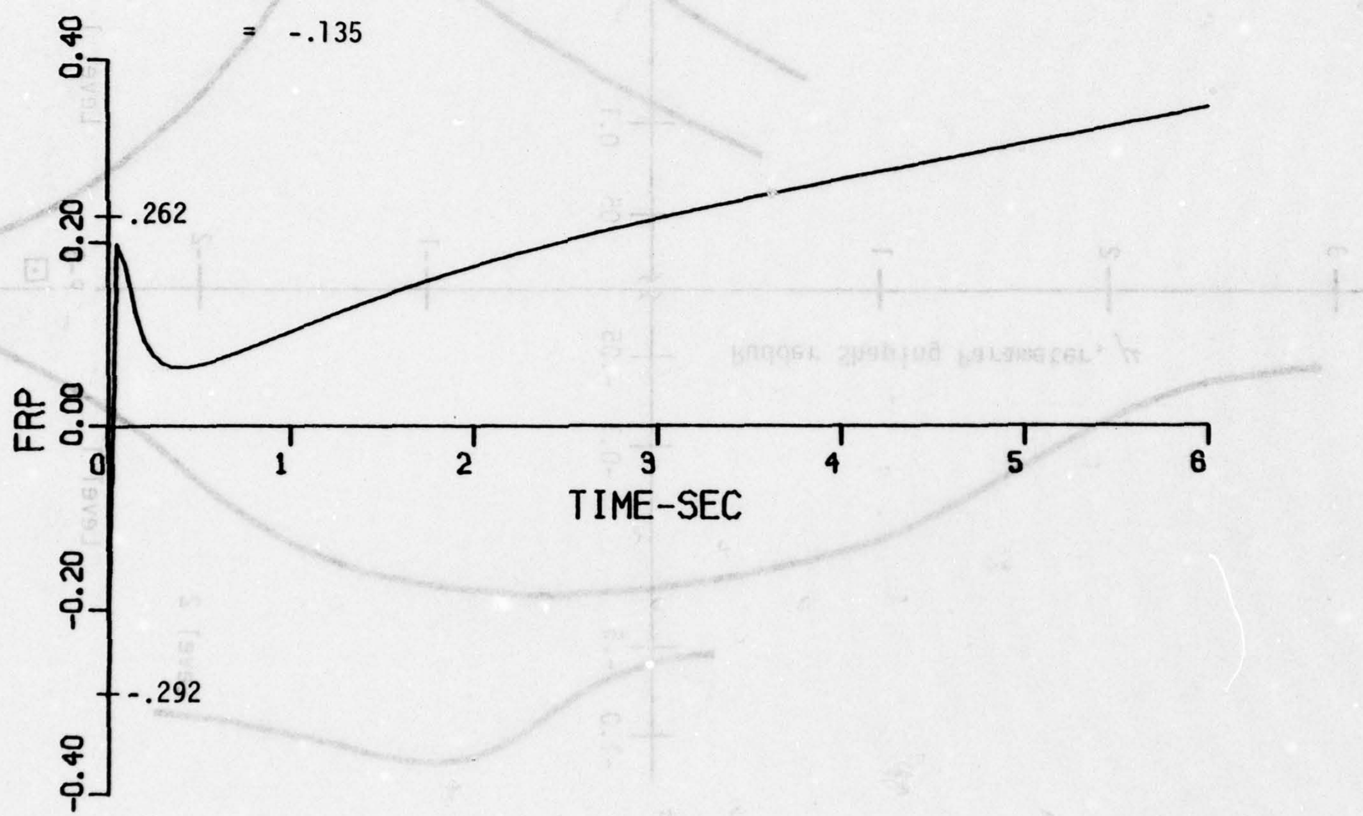


Figure 8 RUDDER PEDAL CROSSFEED FOR YF-16

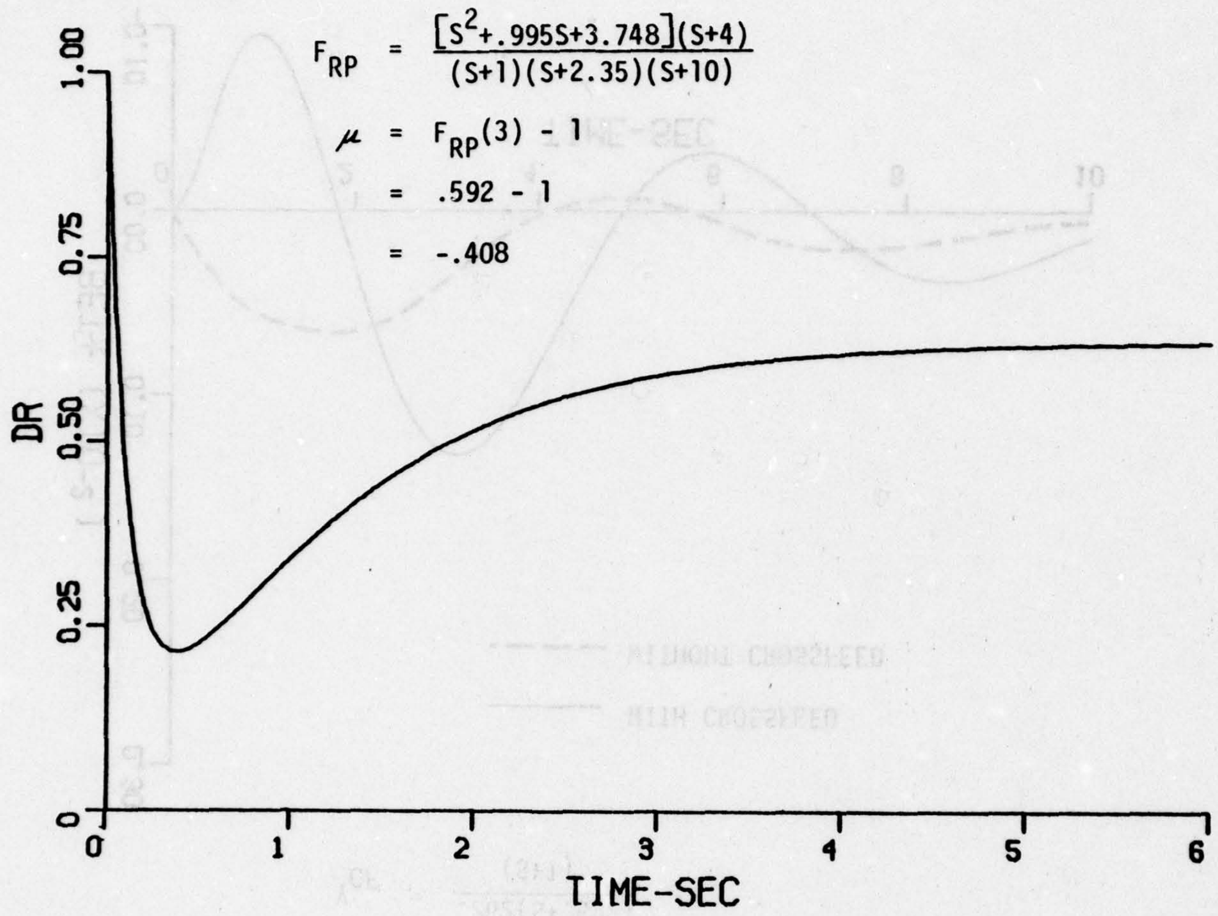


Figure 9 SIMPLIFIED RUDDER PEDAL CROSSFEED FOR YF-16

$$\dot{\beta} = \left[\frac{\dot{\beta}}{\delta_{AS}} + Y_{CF} \frac{\dot{\beta}}{\delta_{RP}} \right] \delta_{AS_C}$$

$$Y_{CF} = \frac{.262(S+.592)}{(S+1)}$$

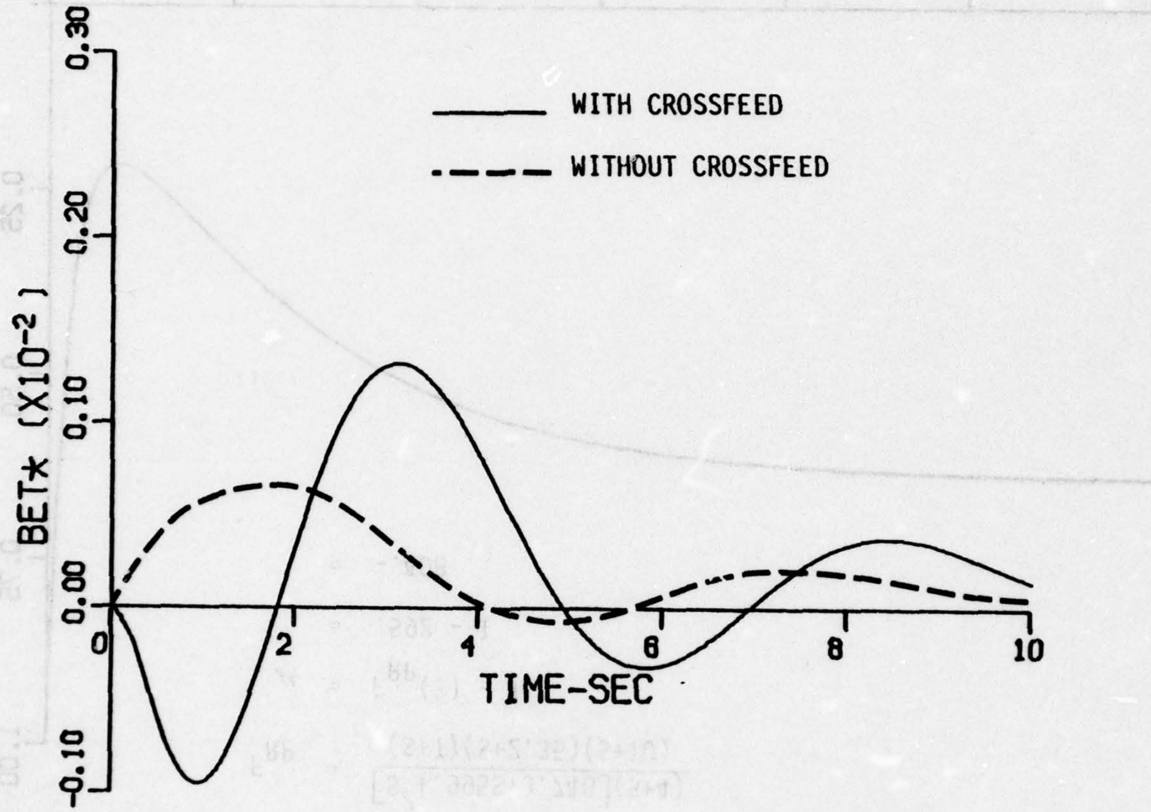
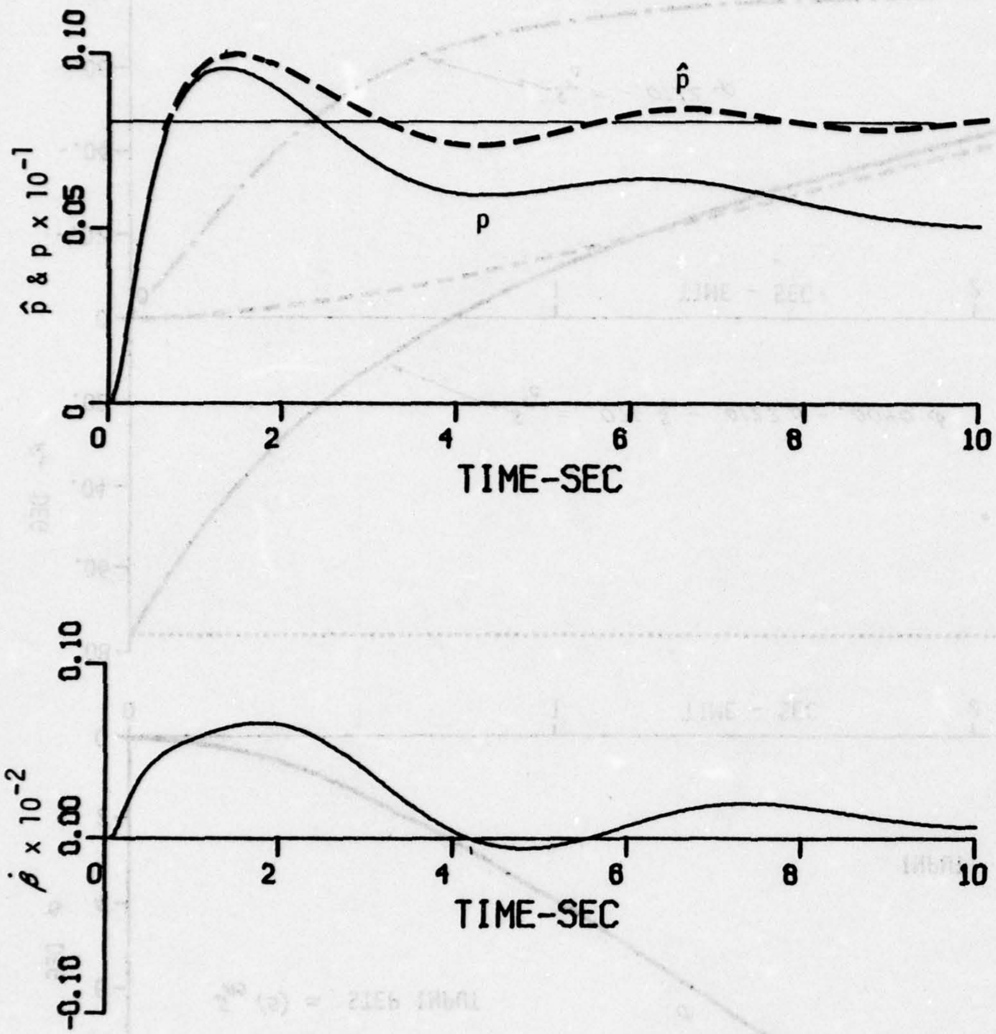


Figure 10 SIDESLIP RATE RESPONSE FOR YF-16 USING CROSSFEED IMPLIED BY μ = -.408



$$\frac{\hat{p}_{OS}}{\hat{p}_1} = \frac{.010 + .00828 - 2(.00739)}{2(.010)}$$

$$= .176$$

$$\psi_{p_{STEP}} = -360\left[\frac{6.3}{5.3} + 1 - 2\right] - \sin^{-1} .25$$

$$= -83^\circ$$

$$\frac{1}{\omega_d}\ \frac{V_T}{g}\ \frac{\Delta\dot{\beta}_{MAX}}{\hat{p}_1} = \frac{1}{1.22}\ \frac{223}{32.2}\ \frac{7.20}{10.0}\ \frac{10^{-4}}{10^{-3}}$$

$$= .409$$

$$\psi_{\dot{\beta}_{STEP}} = -360\left[\frac{7.4}{5.3} + 1 - 2\right] - \sin^{-1} .25$$

$$= -158^\circ$$

Figure 11 ROLL RATE AND SIDESLIP RATE RESPONSES TO STEP AILERON FORCE, YF-16

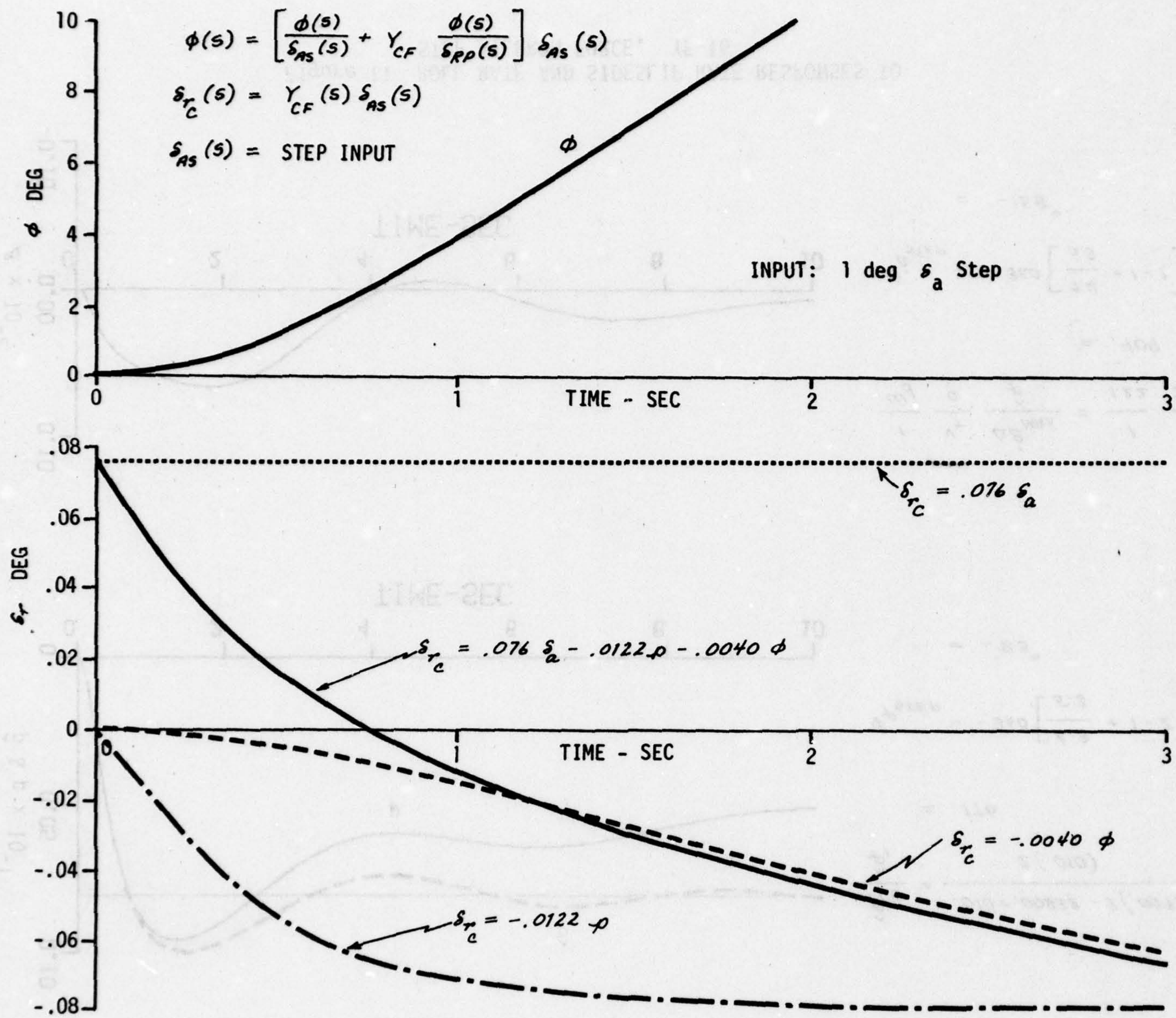


Figure 12 APPROXIMATION OF CROSSFEED TIME HISTORY

Appendix A

REVISED LATERAL-DIRECTIONAL COUPLING REQUIREMENTS PROPOSED BY CALSPAN

3.3.2.2 Bank angle oscillations. The value of the parameter $\hat{\phi}_{osc}/\hat{\phi}_1$ following a rudder-pedals-free impulse aileron control command shall be within the limits in Figure A4 for Levels 1 and 2. The impulse shall be as abrupt as practical within the strength limits of the pilot and the rate limits of the aileron control system. For all Levels, the change in bank angle shall always be in the direction of the aileron control command.

3.3.2.2.1 Additional roll rate requirement for small step inputs. The value of the parameter $\hat{p}_{osc}/\hat{p}_1$ following a rudder-pedals-fixed step aileron command shall be within the limits shown on Figure A4 for Levels 1 and 2. This requirement applies for step aileron control commands up to the magnitude which causes a 60 degree bank angle change in 1.7 T_d seconds. For all Levels, the change in bank angle shall always be in the direction of the aileron control command.

3.3.2.3 Sideslip excursions. Following a rudder-pedals-fixed step aileron control command, the sideslip increment, $\Delta\beta$, shall be less than the values specified herein. The aileron command shall be held fixed until the bank angle has changed at least 90 degrees.

Level	Flight Phase Category	$\Delta\beta$
1	A	6 degrees
	B&C	10 degrees
2	All	15 degrees

3.3.2.3.1 Additional sideslip requirement. The amount of (sideslip) (rate of change of sideslip) following a rudder-pedals(-free) (-fixed) (impulse) (step) aileron control command shall be within the limits shown on Figure A5 for Levels 1 and 2. The impulse shall be as abrupt as practical within the strength limits of the pilot and the rate limits of the aileron control system. The requirement shall apply for step aileron control commands up to the magnitude which causes a 60-degree bank angle change within T_d seconds.

Lateral-Directional Parameter Definitions

ϕ — bank angle measured in the $y_s - z_s$ plane, between the y_s - stability axis and the horizontal

p — roll rate about the x_s stability axis

$\hat{\phi}(t) = \phi(t) + K_{s_\phi}(1 - e^{-\lambda_s t})\delta_{AS_{IMPULSE}}$ Bank angle response to an impulse aileron input with the spiral component subtracted.

$\hat{p}(t) = p(t) + K_{s_p}(1 - e^{-\lambda_s t})\delta_{AS_{STEP}}$ Roll rate response to a step aileron input with the spiral component subtracted.

$\frac{\hat{p}_{osc}}{\hat{p}_1}$ — a measure of the ratio of the oscillatory component of roll rate to the roll rate at the first peak following a rudder-pedal-fixed aileron control command.

$$\frac{\hat{p}_{osc}}{\hat{p}_1} = \frac{\frac{1}{2}\left[(\hat{p}_1 - \hat{p}_2) + (\hat{p}_3 - \hat{p}_2)\right]}{\hat{p}_1}$$

where $\hat{p}_1$, $\hat{p}_2$ and $\hat{p}_3$ are roll rates at the first, second and third peaks, respectively (Figure 126).

$\frac{\hat{\phi}_{osc}}{\hat{\phi}_1}$ — a measure of the ratio of the oscillatory component of bank angle to the bank angle at the first peak following a rudder-pedal-free impulse aileron control command.

$$\frac{\hat{\phi}_{osc}}{\hat{\phi}_1} = \frac{\frac{1}{2}\left[(\hat{\phi}_1 - \hat{\phi}_2) + (\hat{\phi}_3 - \hat{\phi}_2)\right]}{\hat{\phi}_1}$$

where $\hat{\phi}_1$, $\hat{\phi}_2$, and $\hat{\phi}_3$, are bank angles at the first second, and third peaks, respectively. (Figure 126)

$\Delta\beta_{max}$ — maximum sideslip excursion at the c.g., occurring within 1.2 T_d sec following a rudder-pedals-free impulse aileron control command. Usually the peak to peak amplitude (Figure 127)

$\Delta\dot{\beta}_{max}$ maximum sideslip rate excursion at the c.g. occurring within 1.2 τ_d sec following a rudder-pedals-fixed step aileron control command (Figure 127)

proposed flying qualities parameter to limit sideslip excited by roll commands.

$$\frac{\left|\Delta\dot{\beta}_{max}\right|_{t<1.2\tau_d}}{\omega_d\left[\frac{g}{V_T}\hat{\Phi}_1\right]}$$

$\psi_{x_{\substack{IMPULSE\\OF\ STEP}}}$ phase angle expressed as a lag for a cosine representation of the Dutch roll oscillation in the (x) response resulting from an aileron impulse or an aileron step input

$$\psi_{x_{\substack{IMPULSE\\OF\ STEP}}} = -360\left[\frac{t_{n_x}}{\tau_d}+1-n\right]-\sin^{-1}\zeta_d \ deg$$

with n as in t_{n_x}, below. Where (x) is bank angle, roll rate, yaw rate, yaw acceleration, sideslip or sideslip rate response with the spiral residue subtracted (Figures 126, 127)

$\psi_{x_{\substack{IMPULSE\\OF\ STEP}}}$ phase angle expressed as a lag for a cosine representation of the Dutch roll oscillation in the (x) response resulting from an impulse or step aileron-control command

$$\psi_{x_{\substack{IMPULSE\\OF\ STEP}}} = \sum\theta_z - \sum\theta_p$$

where:

θ_z angles of line segments from zeros of $\frac{x(s)}{\delta_{AS}(s)}$ transfer function numerator to the positive conjugate Dutch roll root, for δ_{AS} impulse or step.

θ_p angles of line segments from poles of $\dfrac{x(s)}{\delta_{AS}(s)}$ transfer function denominator to the positive conjugate Dutch roll root, for δ_{AS} impulse or step.

t_{n_x} time for the Dutch roll oscillation in the (x) response to reach the n^{th} local maximum for a right step or impulse aileron-control command, or the n^{th} local minimum for a left command. In the event a step command is employed, the control shall be moved as abruptly as practical and, for purposes of this definition, initial time, t_o , shall be defined as the instant the cockpit control deflection passes through half the amplitude of the commanded value. For pulse inputs, time shall be measured from the point halfway through the duration of the pulse. Only peaks occurring after $t = 3\tau_R$ should be used to avoid distortion resulting from the roll mode.

K_{s_x} Residue of spiral root in the (x) time history response to aileron impulse or aileron step input.

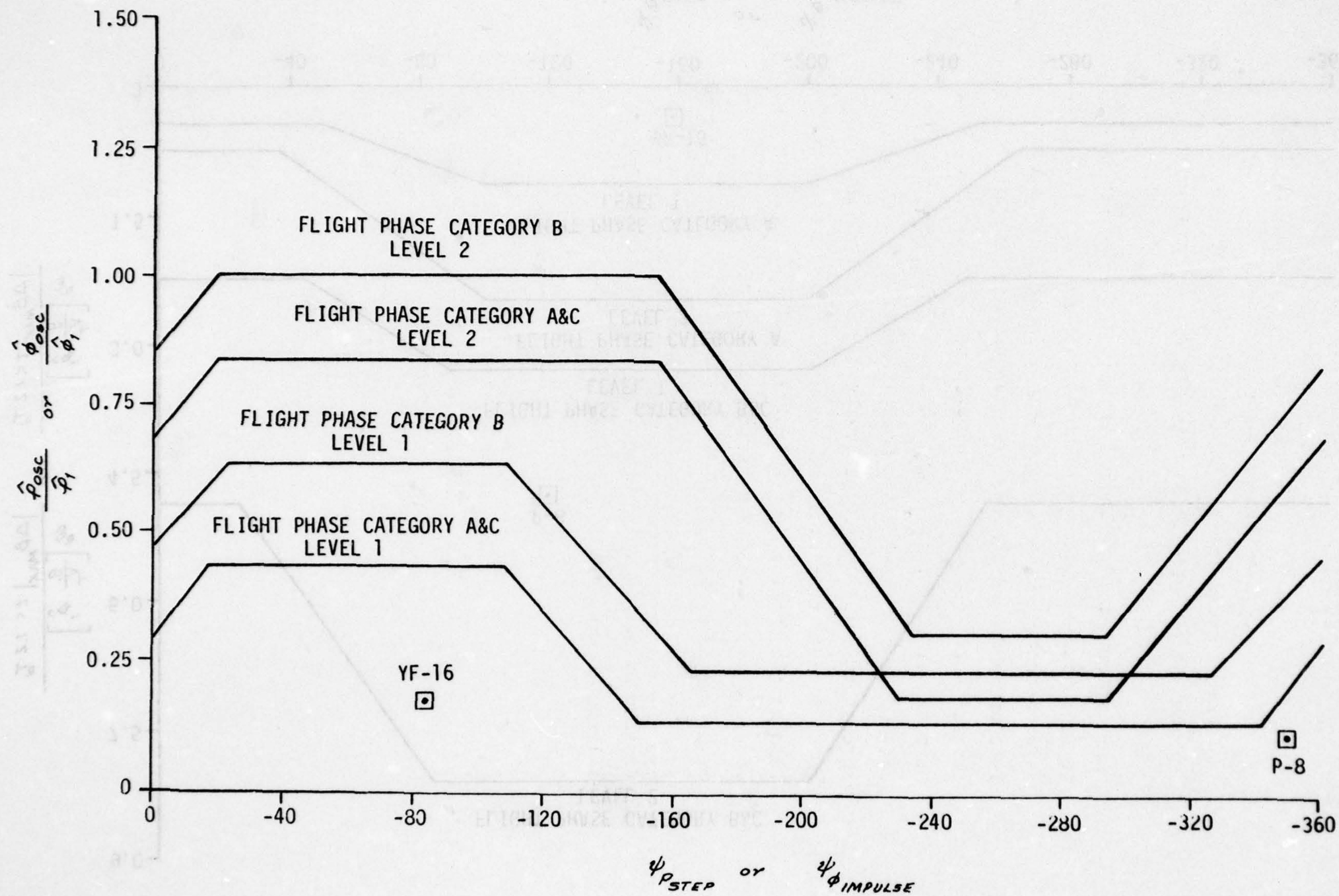


Figure A4 ROLL RATE AND BANK ANGLE OSCILLATION LIMITATIONS

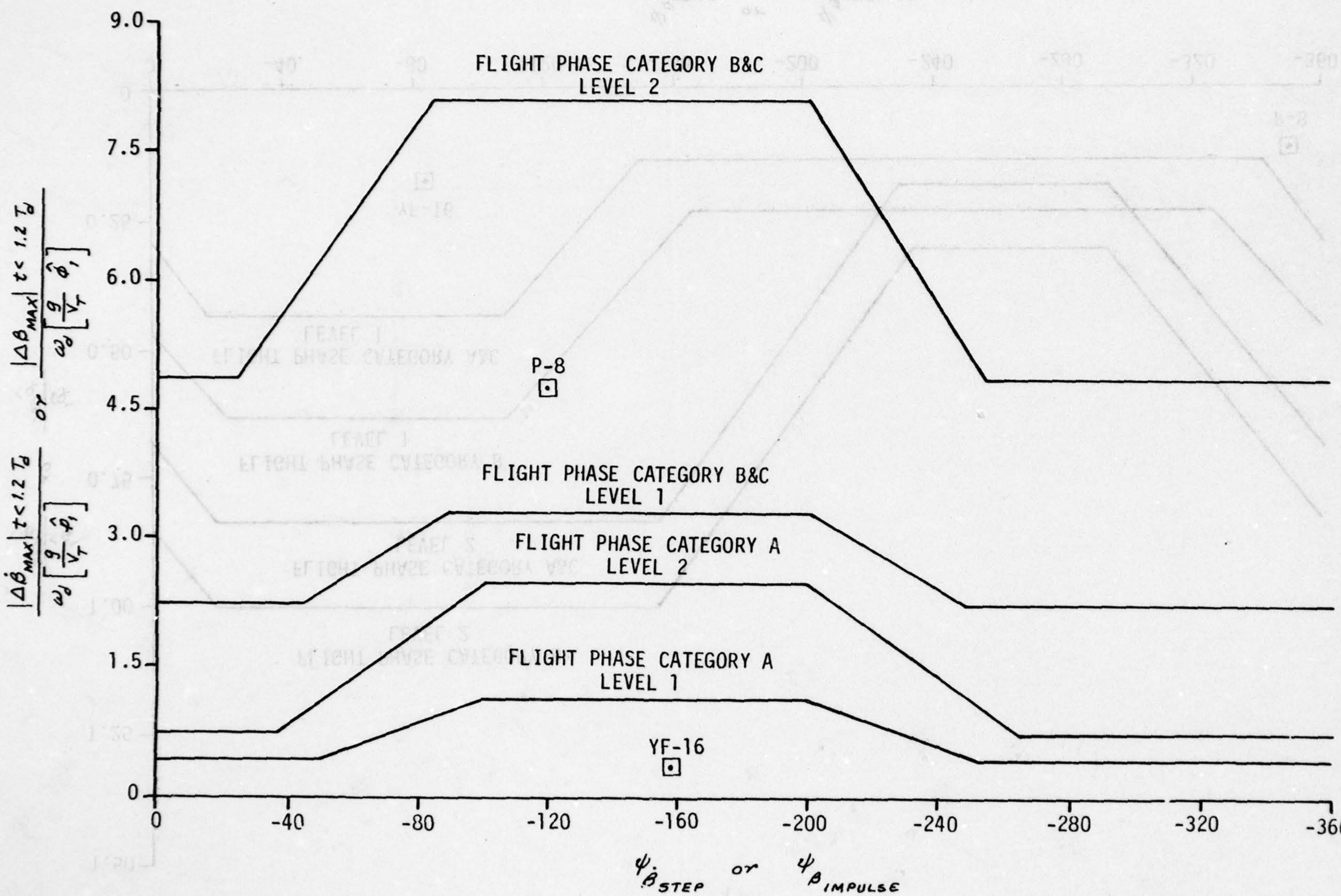


Figure A5 SIDESLIP EXCURSION LIMITATIONS

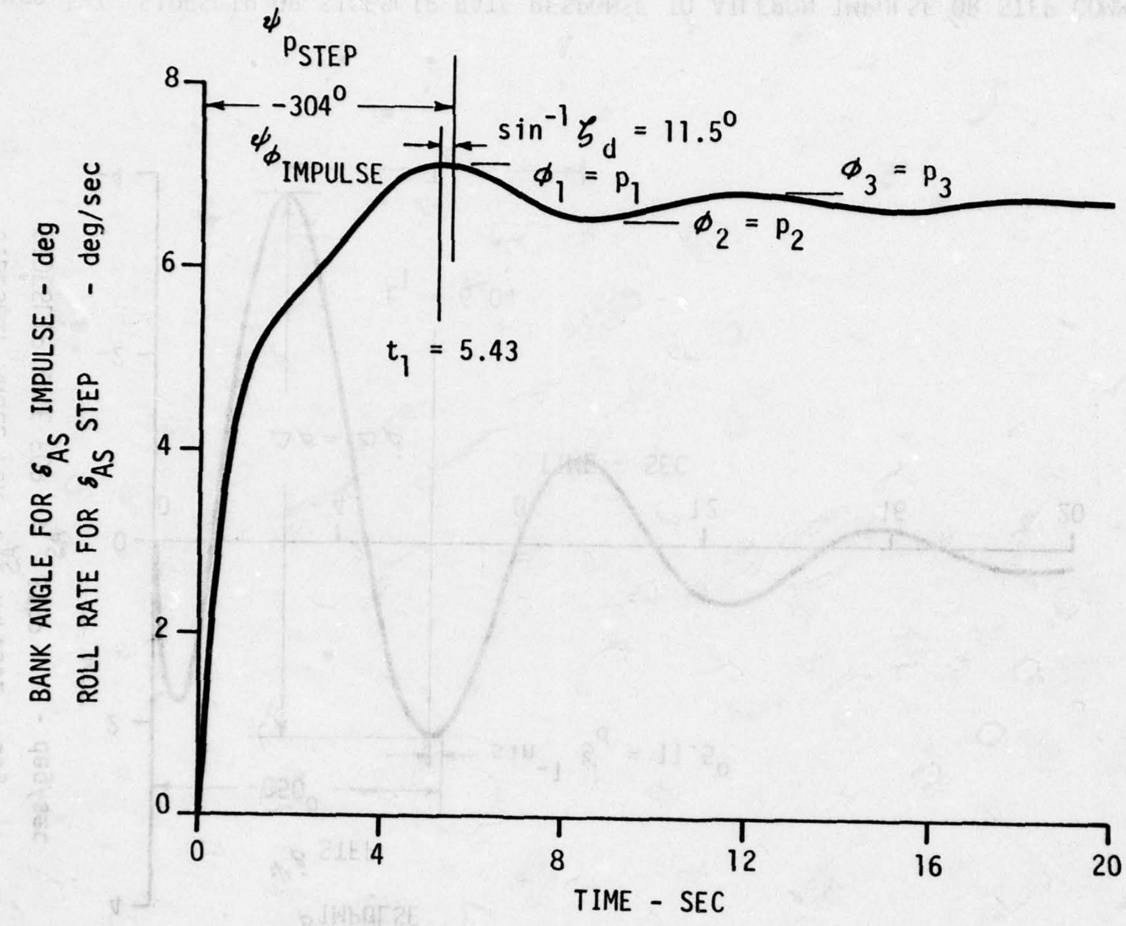


Figure 126 BANK ANGLE OR ROLL RATE RESPONSE TO AILERON IMPULSE OR STEP COMMAND

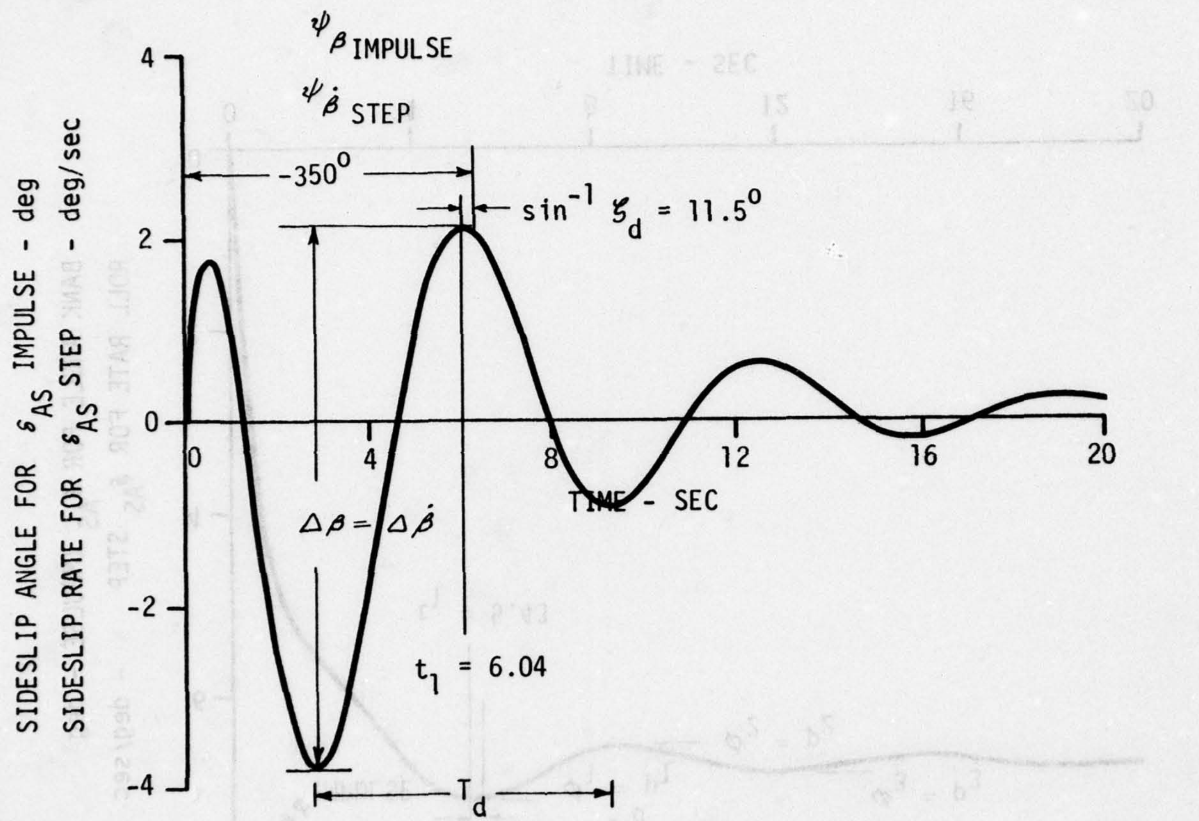


Figure 127 SIDESLIP OR SIDESLIP RATE RESPONSE TO AILERON IMPULSE OR STEP COMMAND

Roger Hoh, STI: 1. Your primary argument with the μ parameter appears to be the simplification of the crossfeed. We no longer recommend that high or low frequency roots be removed and that the crossfeed (with unity high frequency gain) be used in its raw form to calculate $\delta_r = (3)$. How does this affect your objections?

2. You show a rudder time history which is non-monotonic. This is an important piece of data because such large departures from the assumed monotonic shape are expected to result in poor pilot ratings. What was the pilot rating for that configuration? (Original YF-16) What was the value of $\frac{N_{\delta\omega}}{L_{\delta\omega}}$ for the original YF-16? (Obtain $N_{\delta\omega}$ as high frequency gain of $(N^{P}_{\delta\omega})_{aug}$ respectively).

3. What was $\frac{N_{\delta\omega}}{L_{\delta\omega}}$ on the configuration that met both criteria and was bad?

Answer: 1. If the rules for measuring $\delta_r(3)$ are different from those published in the STI AIAA article and the AFFDL report, then why didn't Mr. Hoh describe these new rules in his presentation before lunch today? I believe I have shown that cases can exist for which the μ parameter as defined in the written reports, implies a crossfeed time history that is not at all shaped like the rudder time history required to restrain sideslip to be zero and in fact will cause larger sideslip excursions than would occur if no rudder was used.

It is not clear to me what the new measurement rule is; therefore, I am not prepared to comment on its utility.

Examination of the stick to rudder crossfeed required to constrain sideslip is certainly worthwhile, and as I indicated in my paper, the rudder time history can be interpreted in terms of crossfeed and feedback gains δ_r/δ_w, δ_r/p and δ_r/ϕ which can be used to diagnose the coordination problem of a given airplane. These gains are also effective augmentation loops. Whether or not an adequate flying qualities criterion can be developed which uses rudder amplitude conformation at only one or two instants in time is not clear to me. The STI approach does not seem to treat cases where the pilot chooses not to use the rudder.

2. I do not know what pilot rating the YF-16 has been given for the flight condition the transfer function represents. The ratings from the NT-33A simulation should not be used because there was no attempt to mechanize the details of the YF-16 lateral directional control system. The ratio of high frequency coefficients in r/δ_w and p/δ_w transfer functions is (-.111).

3. For configuration P-8 the value of $N'_{\delta w}/L'_{\delta w}$ was t.035.

Dwight Schaeffor, Boeing: Do the feedbacks to rudder employed to make $\beta/\delta\omega = 0$ adversely affect steady turn (hands off) coordination or response to turbulence?

Answer: The feedbacks δ_r/p and δ_r/ϕ and the crossfeed δ_r/δ_w are designed to help keep sideslip zero in steady turns and rolling maneuvers. The δ_r/ϕ feedback will effect the spiral root and it may be necessary to use yaw rate feedback to the aileron $\delta a/r$ to keep the spiral root neutral or stable. The feedback signals are inertial, therefore the airplane tends to be stabilized and to reject turbulence upsets.

SECTION VI

WORKING SESSIONS ON THE PROPOSED REVISION

WORKING SESSION ON THE PROPOSED REVISION

GROUP 1: Discussion of the Proposed Revision from a Design Standpoint (Session Moderator: Tim Sweeney)

SUMMARY

In this session, the goal was to determine the effects of the proposed revision on the designer in his efforts to develop a vehicle with good flying qualities. The goal as such was not met, but a consensus was reached on a number of elements in the proposed revision. The changes to the roll response requirements, the transients and the deletion of references to surface displacement were all considered improvements. The addition of wording about sidestick controls was not considered to be of any help and the proposed change to "Levels of flying qualities," paragraph 1.5, was not considered a good change and many were concerned about the potential for confusion if it were implemented. Much of the rest of the discussion centered on the philosophy of specifications and how they are applied; the remaining problem areas not dealt with in the proposed change and on misinterpretations or misunderstandings of the specification and proposed changes.

The opinion was offered that the specification was merely a listing of lessons learned and that it provided little design guidance. This opinion is fairly accurate and it was pointed out that the "Mil Prime" format should clarify this once and for all, since they are to have sections in the handbook tabulating specific lessons learned and are explicitly not intended to provide design guidance. It was agreed that design approaches are, and should remain, the contractors' prerogative in systems development. A question was raised concerning how the specification was used by the government and by the contractors. Most agreed that the use varies with the particular procurement, whether new or off-the-shelf for example, and with the individual contractor. It was pointed out that in preliminary design, performance, not flying qualities is foremost and therefore flying qualities considerations have a second order effect on the overall configuration. Another question was raised concerning whether the specification applies to Class I aircraft since attempts at compliance would undoubtedly drive costs up. That situation may be academic however, since recent Class I aircraft acquisitions have been of off-the-shelf models, but it is a valid point that strict implementation of MIL-F-8785B could represent an unwarranted expense and technical complexity for a simple aircraft. The "Mil Prime" format offers a real opportunity to deal with this situation in an efficient manner, simplifying the task of specification compliance (at the option of the buyer) for systems where the risk is low.

A great deal of time was spent on the problems with MIL-F-8785B which the proposed change did not address. On some of these, there was a consensus in the group that improvement should be considered for the proposed amendment. Examples of these are: that some of the requirements on modal parameters are very difficult and may be unnecessary to meet at very high altitudes (60000^+ feet); that the upset transients (at least the excursions) more properly apply to category A flight phases only; that the terminology we apply to the various envelopes creates communication problems especially with the operational people and finally, that the gradient change at high load factors for structural protection was not allowed by the wording of the proposed revision.

In addition to the above, there were a number of complaints by individuals on various aspects of the specification. One person felt that all qualitative requirements should be deleted since they constituted open-ended compliance situations which had proven costly in his experience. Another decried the extensive parameterization in the spec and asked if task-oriented requirements could be developed to reduce it. This was countered by an individual who felt that pilot-in-the-loop requirements could not be made to work in a specification. It was also mentioned that the critical pilot-in-the-loop task may not be known until after the vehicle has been flown and tactics developed. This was the experience on the A-10 aircraft. One group member added that the same task may necessitate different requirements for different parts of the flight envelope. The question of configurations to which the level one requirements apply was brought up again. The specific instance involves a pilot-assist mode which is required to achieve level one performance in certain flight phases. The question centers around the fact that since the mode is a selectable configuration paragraph 3.1.5 could be interpreted as requiring compliance with level one spec values with it ON and OFF. It was requested that, as a minimum, clarification material be added to the background document.

There was some confusion concerning the proposed change to the section on permissible flight envelope where some individuals thought that this envelope could be smaller than the service or operational envelopes. It was shown that the wording of 3.1.10.3.3 (which was not changed) strongly implies that permissible flight envelope boundaries are outside those of the service flight envelope and that this is the intended relationship.

This working group was attended by a good mix of industry and government personnel. The above summary represents the workshop findings and recommendations within the limits of memory and note taking capability. A list of attendees and their affiliations is attached.

1 Atch
List of Attendees

ATTENDEES

NAME	AFFILIATION
R. J. Woodcock	AFFDL/FGC
T. D. Lewis	AFFDL/FGL
J. T. Clay	Beech Aircraft Corp
Paul P. Shipley	Rockwell Space
Donald E. Johnston	Systems Technology Inc
C. F. Anderson	Lockheed Calif Co
Ralph Smith	SRL
J. E. Buckley	McAIR
J. E. Kremowski	Fairchild Republic Co
Ray Kostanty	Northrop Corp
Ron Anderson	AFFDL/FGC
Frank Carlson	Boeing
Brian W. VanVliet	AFFDL/FGC
Marty Moul	NASA/LRC

Workshop Discussion of the Proposed Revision from a Flight Test and Operations Viewpoint

Moderator: Capt. Roy Martin, USAF Flight Test Center

Introduction:

The flight test group's discussion was centered around a concern with the requirement to verify the ability of a given real pilot/airplane/flight control system to meet the specification of MIL-F-8785B and to ultimately accomplish given mission tasks. More specifically the requirements as outlined in the Revised Version of MIL-F-8785B were addressed by this group.

The following discussions, conclusions, and recommendations were the consensus of the flight test discussion group.

Paragraph 1.5 Levels of Flying Qualities

Discussion: The entire group felt that the incorporation of the atmospheric disturbance criteria into the "Level" definitions is unacceptable. Whereas, the existence of a degraded environment may well degrade the observed performance or increase the necessary compensation it is impossible to define the external disturbance with existing flight test instrumentation. To effectively test against the proposed paragraph it would require a continual metric of atmospheric state to evaluate which "level 1" applies. The level of flying qualities relates to the desired task, i.e., a low-level ground attack aircraft may well encounter inordinate turbulence (as defined in the proposed revision) with a much higher probability than 10^{-3} and still require clearly adequate flying qualities for this mission flight phase. That is the required operating environment may normally include airmass disturbances of this intensity.

If reference to degraded Flying Qualities with atmospheric disturbance is to be made it should not be done by changing the level definitions but possibly be treated as a failure state, i.e. moderate turbulence would allow satisfactory compliance with the specification by meeting level 2 requirements at the discretion of the procuring agency.

Conclusion: The concept of the proposed revision to paragraph 1.5 is unacceptable to the entire group. Any automatic degradation of F.Q. levels with atmospheric disturbance as part of the specification would be opposed by all agencies represented at this session.

Recommendation: Leave paragraph 1.5 as presently written.

Paragraph 3.1.1. Operational Missions

Discussion: Clarification necessary and correct

Recommendation: Publish as revised

Paragraph 3.2.2.1.3 Higher Order Dynamic Systems

Discussion: Although there is a definite need to insure that higher order dynamic systems are acknowledged and that adequate and safe performance is specified, the group is not convinced that the equivalent system approach should be adopted at this time.

Additional research is necessary to better define the boundaries and to produce lateral directional guidance. At the present time, flight test techniques are available to generate equivalent systems parameters although verification is still required.

Conclusion: Additional research is required but some acknowledgment of higher order systems requiring verification of safety and performance but with no legislation of technique.

Paragraph 3.2.1.1 Longitudinal Static Stability

Discussion: Considering the statement, "In no event shall there be more than one unstable mode of motion, whether it be aperiodic or oscillatory," if by modes of motion this paragraph implies the classical dynamic modes of motion (i.e. Short Period, Phugoid, Dutch Roll, Roll, Spiral Mode), then an interpretation could imply that an unstable spiral mode would require all other modes to be stable. If the paragraph is meant to refer only to the longitudinal modes (i.e. Short Period and Phugoid) then an interpretation could imply that an unstable Phugoid would require a stable Short Period. This criterion may be more restrictive than is warranted since divergent phugoids with long periods may be acceptable when coupled with "slowly" diverging Short Period modes.

Conclusion: Clarification of this statement in the paragraph is warranted.

Considering the statement, "In no event shall its time to double amplitude be less than 6 seconds," it should be clarified as to how the 6 sec criteria was obtained. If this 6 sec criteria was determined based solely on pilot evaluations from closed loop high gain tasks (i.e. landing tasks) representative of flight phase category A and C, then further investigation is warranted to insure acceptability of the 6 sec criteria during the Category B flight phase tasks (i.e. long range cruise while operating with the Level 3 failure state).

AD-A066 493 AIR FORCE FLIGHT DYNAMICS LAB WRIGHT-PATTERSON AFB OHIO F/G 1/2
PROCEEDINGS OF AFFDL FLYING QUALITIES SYMPOSIUM HELD AT WRIGHT --ETC(U)
DEC 78 G T BLACK, D J MOORHOUSE, R J WOODCOCK

UNCLASSIFIED AFFDL-TR-78-171 NL

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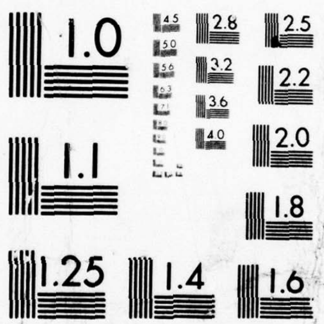
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In addition, the statement, "In the presence of one or more other Level 3 flying qualities no static longitudinal instability will be permitted unless....", there is confusion as to which other flying qualities are implied. Does this refer only to other longitudinal paragraph requirements or the any other paragraph requirement throughout the MIL SPEC in which the aircraft can meet only Level 3 flying qualities?

Conclusion: Clarification of these statements in the paragraph is warranted.

Recommendation:

1. Clarification of wording of the specific paragraph statements identified is recommended.
2. Recommend that the minimum 6 sec time to double amplitude requirement be verified as applicable to Flight Phase Category B tasks (i.e., consider the consequences of a pilot on a long overwater mission whose Level 3 failure occurs with over two hours to the nearest suitable landing site. Is the longitudinal instability defined by the 6 sec to double amplitude too difficult to control for over 2 consecutive hours of flight?).

Paragraph 3.2.2.2.1 Control Forces in Maneuvering Flight

Discussion: Considering the statement, "...A departure from linearity resulting in a local gradient which differs from the average gradient for the maneuver by more than 50 percent is considered excessive."

Conclusion: The average gradient of Fs/n_z is not defined anywhere in the MIL SPEC.

Recommendation: Recommend that a definition of average gradient of Fs/n_z be defined. Consideration should be given to adapting the definition as outlined in FDL report TR 71-134.

Paragraph 3.2.2.2.2. Control motions in maneuvering flight

Discussion: The average side stick control motion requirement of .5 pound per degree for levels 1 and 2 is in error. Reference 36* outlines a minimum side stick gradient of .5 degree per pound; however, recent flight test data from FDL funded tests refute this sidestick gradient criteria. The data base at this time is insufficient to establish a specified side stick control motion gradient.

* Hall, G. Warren and Smith, R.E., "Flight Investigation of Fighter Sidestick Force-Deflection Characteristics", AFFDL-TR-75-39, May 1975.

Recommendation:

1. Delete the .5 pound per degree for levels 1 and 2 requirement and replace this criteria with such wording that side stick control motion gradients are to be determined.
2. Further testing to specify acceptable sidestick control motion gradients is recommended.

Paragraph 3.2.2.4 Longitudinal pilot-induced oscillations

Discussion: Consider the statement, "The following requirements shall be met when the pitch control is pumped sinusoidally for all amplitudes within the structural limits of the airframe at frequencies between 1 and 10 radian/second."

There is no way to adequately or safely flight test the requirement as stated in this paragraph. The words "all amplitudes" imply maximum stick deflections must be tested and "within the structural limits" could be interpreted to mean that this test must be accomplished at all g levels attainable in the aircraft. Irregardless, max amplitude pitch control sinusoidal pumping is not recommended for safety of flight reasons.

The requirements for longitudinal PIO as stated in paragraph 3.2.2.4.3 Control System Phase Lag, utilize the criteria as described in Reference 71.* This reference is considered adequate as a longitudinal PIO Spec at this time.

Recommendation: Recommend that the following statement be deleted from paragraph 3.2.2.4, "The following requirements shall be met when the pitch control is pumped sinusoidally for all amplitudes within the structural limits of the airframe at frequencies between 1 and 10 radian/second."

Paragraph 3.3.4.1 Roll performance for Class IV airplanes

Discussion: This requirement states that "Roll performance for Class IV airplanes is specified over the following ranges of airspeeds in 1-g flight." This paragraph implies that all roll performance tests must be accomplished in 1 g flight. This precludes using a bank-to-bank test method where precisely 1 g may not be maintained throughout duration of the test. Irregardless of the test method employed, precise 1 g control can probably not be maintained throughout the roll performance tests as required by Table IX b.

* Smith, Ralph H., "A Theory for Longitudinal, Short-Period Pilot-Induced Oscillation", AFFDL-TR-77-57, June 1977.

Recommendation: Recommend that the words "in 1-g flight" be deleted from the last sentence of paragraph 3.3.4.1 (Revised).

Paragraph 3.4.11 <u>Control Margin</u>

Discussion: Considering the statement, "Control authority and rate margins shall be sufficient to assure safety in regions of control-surface-instability throughout the combined range of all attainable angles of attack (both positive and negative) and sideslip. This requirement applies to the prevention of loss of control and to recovery from any situation for all maneuvering."
Where is control authority defined in the MIL SPEC?
Where are rate margins defined?
What does "sufficient to assure safety" mean?
Who decides what is safe?
Are control authority and rate margins to be sufficient to assure safety at all attainable trimmed angles of attack and sideslip or all attainable angles of attack and sideslip? (E.g. an F-16 type of aircraft pointed straight up until airspeed equals zero then begins to backslide may achieve some dramatic α's and β's. Are control authority and rate margins to be required for this condition?)

Recommendation: Recommend that the "words" of this paragraph as highlighted be more clearly defined.

Workshop Discussion of the Proposed Revision from an Analytical Concepts and Methods Viewpoint

Moderator: David J. Moorhouse, AFFDL/FGC

The discussion opened by acknowledging that the proposed revision of the Levels of Flying Qualities had already produced disagreement. The group decided, therefore, to consider other revisions first.

Longitudinal static stability (3.2.1.1)

There was both support for and opposition to the proposed allowance of Level 3 instability. The requirement was noted to allow an unstable airframe but not require it, that would remain a design option. The discussion then centered on the proposed value for time to double amplitude. The consensus was that 6 secs is adequate, even conservative, if the pilot is tightly controlling the aircraft without side tasks. For Flight Phases such as cruise, however, (or "night IFR glideslope intercept") it was felt that 6 secs is too short when the pilot may be occupied with side tasks such as radio transmission, navigation, etc.

The judgment was to reassess the proposed Level 3 requirement - possibly as a function of Flight Phase and airplane Class. There were also requests to clarify the intent of the proposed: "In no event shall there be more than one unstable mode of motion."

Higher order dynamic systems (3.2.2.1.3)

This subject also generated much discussion but no consensus. There was a general feeling that the frequency domain has meaning to designers, while time domain envelope criteria are not discriminating (in particular they would not screen out high frequency effects). Discussion on the equivalent system approach, per se, raised a variety of points:

1) all modal requirements should be equivalent

2) it is appealing to have all the parameters related, as in the proposed revision
3) there is a high (supersonic) speed problem with pitch attitude tracking
4) other transfer functions need to be considered
5) should the phugoid be included with the short period?
6) specify what is really desired rather than requirements on modal parameters.

The discussion was lively but the only consensus was that more work is required.

Lateral-directional requirements (3.3)

Hodgkinson of McAir again advocated the use of equivalent parameters for all the modal requirements. Specific comments were

1) the required Level 1 damping ratios are too low; since this is frequently provided by augmentation increasing the requirement should not be a major problem
2) the required $\zeta\omega$ values are too stringent for large aircraft because of the low frequencies involved
3) a new requirement was proposed - "There shall be no objectionable coupling between pitch motions and the Dutch roll mode.

There was, moreover, even stronger unanimity that the lateral-directional requirements need improvement.

Roll control requirements (3.4)

Again, there was a suggestion to specify requirements on response to pilot input rather than modal requirements. As part of this discussion, guidance was requested in measuring roll mode time constant for highly augmented airplanes. The proposed revisions were accepted with the discussion concentrating on rudder-pedal-induced rolls (3.3.4.5). The concensus seemed to be that the requirement should be retained in a form to preclude adverse rudder-pedal-induced rolls.

Failure transients (3.5.5.1)

From the discussions of this section, the consensus was that the allowed excursions are very stringent. A strong majority suggested that the numerical requirements could be deleted in favor of the qualitative requirement in 3.5.5 (a lone voice from ASD dissented).

Transfer to alternate control modes (3.5.6.1)

In contrast to the preceding section, the allowed transfer transients were felt to be both appropriate and valid. In terms of clarification, it was felt that this requirement should not apply to switching off an augmentation system in training for failure states.

SECTION VII

SESSION 4: SPECIAL PROBLEMS 2

AN EVALUATION OF SIDESTICK FORCE/DEFLECTION CHARACTERISTICS ON AIRCRAFT HANDLING QUALITIES

Stephen B. Smith
Charles M. Miller, Capt, USAF
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Edwards Air Force Base, California

ABSTRACT

The growing popularity and acceptance of aircraft sidestick controllers has emphasized the lack of data available to design optimized sidestick configurations. This study was conducted by students of the USAF Test Pilot School using the Calspan NT-33A variable stability aircraft to help expand the data base. The NT-33A was configured with a variable force and motion sidestick controller and was programmed to simulate the handling characteristics of a modern high-performance fighter aircraft. Three separate investigations were conducted in this study.

The tasks evaluated consisted of precision air-to-air tracking of a target aircraft during level and windup turns, gross acquisition and tracking involving large pilot inputs, and landings.

The variables of the investigation were the ratio of stick force to stick deflection and the ratio of stick force to aircraft response (either load factor or roll rate). The influence of sidestick controller first order pre-filter characteristics was investigated by performing the tasks with different pre-filters in each axis for selected force/deflection characteristics.

Pilot comments were obtained for all configurations. Pilot ratings of each task were made using the Cooper-Harper rating scale. From this assemblage of comments and ratings, areas bounding the best and the worst combinations of sidestick force and deflection were defined. This discussion is confined to the air-to-air tracking task.

Stephen B. Smith, Aerospace Engineer, Performance and Flying Qualities Branch, Flight Test Engineering Division

Charles M. Miller, Capt, USAF, Aerospace Engineer, Systems Engineering Branch, Flight Test Engineering Division

Acknowledgement

The authors would like to acknowledge the efforts of Robert Harper, G. Warren Hall, and Michael Parag, engineering pilots, Calspan Corp., who were the safety pilots for the evaluation flights and who provided valuable assistance to the program.

Presented at the Society of Flight Test Engineers National Symposium, October 5, 1978, in Arlington, Texas.

INTRODUCTION

The sidestick controller is a serious contender to the center stick in fighter aircraft design, as evidenced by its introduction into service use in the F-16. The available literature contains very little insight into the nature of desireable sidestick characteristics (i.e., mechanization, force-feel, deflection, harmony, etc) or what influence such factors as control system and aircraft dynamics or class of aircraft have upon these characteristics. Recent experience with sidesticks (references 1 and 2) have borne out the need for a thorough and comprehensive series of investigations of sidestick controller characteristics. In addition, supportive data are required for a quantitative requirement on sidestick controllers in the specification for Flying Qualities of Piloted Aircraft, MIL-F-8785B.

In order to provide this additional data, the USAF Flight Dynamics Laboratory, Wright-Patterson AFB, Ohio, had funded the Calspan Corp, Buffalo, NY, to investigate the influence of force and deflection characteristics on air-to-air tracking and landing task performance (reference 3). To continue this effort, the Laboratory has been sponsoring the USAF Test Pilot School at Edwards AFB, California on a continuing series of sidestick investigations. Three of these test programs, flown and reported on by students of the Test Pilot School (references 4, 5, 6, 7) are the subject of this discussion. These programs made use of the Calspan NT-33A variable stability aircraft equipped with a variable force and deflection sidestick controller to perform the evaluations. The investigations were designed such that each would progress in a logical fashion, building upon the findings and conclusions of the previous one.

TEST PROGRESSION

The first phase in the test series was conducted between 13 May and 3 June, 1977 by students of class 76-B (references 4 and 5) and consisted of 23 flights. The investigation was derived as a follow-on to the progenitor of the series, the Calspan program of reference 3. The objective of this phase was to provide an additional, independent investigation of the same variables considered by Calspan; sidestick force per unit of aircraft response and stick deflection per unit of stick force. An important difference between the Test Pilot School investigation and Calspan's was the task employed in the air-to-air portion of the evaluation. It was a more structured and demanding tracking task than the operationally-oriented task favored by Calspan. It was hoped that this task might be useful in high-lighting tracking deficiencies more clearly.

Phase II of the series was conducted to provide additional information on the effects of the same variables and to extend the ranges of these variables to heavier forces and larger amplitudes. The investigation was conducted between 28 October and 25 November, 1977 by students of Class 77-A (reference 6). Data were obtained on 16 flights.

Phase III was conducted by Class 77-B from 15 May to 9 June, 1978 (reference 7). This phase investigated the influence of an additional variable upon the results obtained earlier. Different longitudinal and lateral control system first order pre-filters were chosen for this study. A range of pre-filter corner frequencies was investigated for two force-deflection configurations. The objective of the evaluation was to determine the preferred pre-filter for each configuration and whether the relative acceptance of two significantly different configurations was effected by the pre-filter values. Data were obtained on 19 flights.

TEST AIRCRAFT DESCRIPTION

The test aircraft, NT-33A USAF S/N 51-4120, is a modified T-33A jet trainer capable of reproducing the dynamic response and control system characteristics of different aircraft. Static and dynamic responses of the basic T-33A were modified by a response feedback, variable stability system which positions the control surfaces through full authority electrohydraulic servos. The normal front cockpit flight controls are disconnected from the NT-33A control system and have been replaced by a variable force and deflection fly-by-wire sidestick controller which was used in this evaluation. Aircraft modification and maintenance were performed by Calspan.

The electrohydraulic variable force and deflection sidestick controller is shown in figure 1. This sidestick is operated with independently variable force gradients and pre-filters in both the pitch and roll axes. The sidestick pivot point for the longitudinal and the lateral axes is the stick base. Control force gradients are achieved through an electrohydraulic system built into the side controller. Aircraft trim is available with a sidestick-mounted trim switch.

The dynamic characteristics of a modern high-performance fighter aircraft were implemented using the NT-33A variable stability system. These characteristics were similar to those used in the Calspan experiment. The test airplane dynamic characteristics are shown in the table below. (The dynamics varied somewhat between phases; these are representative values). Force commands were used in both the lateral and longitudinal axes. Force/response gains were unaffected by changes in feel system force/displacement gradients or pre-filter values.

Parameter		Air-To-Air Dynamics	Approach & Landing Dynamics
n_z/α	g/rad	33	7
ω_{sp}	rad/sec	5.0	2.2
ζ_{sp}	--	0.6	0.5
ω_p	rad/sec	0.09	0.15
ζ_p	--	0.05	0.05
τ_r	sec	0.2	0.5
τ_s	sec	∞	∞
ω_d	rad/sec	3.2	1.2
ζ	--	0.4	0.25
$\|\phi/\beta\|$	--	0.5	3

During the evaluation, a Calspan safety/instructor pilot varied the computer gains through controls located in the rear cockpit and thus could change the dynamics of the aircraft and the control system characteristics during flight. The aircraft was also equipped with a fixed depression gunsight and a gunsight camera.

SIDESTICK CONFIGURATIONS

Air-to-Air Evaluation. The combinations of force-to-aircraft response and deflection per unit of force ratios that were evaluated for the air-to-air tasks in all three phases are shown in the matrix of figure 2. The control force gradients denoted by name in the matrix are also depicted in figures 3 and 4 as plots of force versus aircraft response. The non-linearity of the force/response gradients was designed by Calspan to depict modern fighter aircraft control mechanizations. The definitions applied to each force/response gradient were vague, and as seen in the figures, varied from one phase to the next. The variation of deflection with force was linear in each case, and the numerical values are therefore depicted. For each configuration evaluated, the longitudinal and the lateral force and deflection characteristics were chosen as congruous pairs (i.e., a "heavy" lateral force was used in conjunction with a "heavy" longitudinal force) in order to provide control harmony. The breakout forces in both axes were 0.5 pounds.

Approach and Landing Evaluation. The control force gradients employed for these tasks were approximately one-half those shown in figures 3 and 4. The same deflection gradients were used for both approach and air-to-air tasks.

FLIGHT MANEUVERS AND TEST TECHNIQUES

The maneuvers flown in the series varied somewhat between the three phases. All phases, however, utilized a highly structured tracking task described in detail in reference 8. Phases II and III investigated gross maneuvers as additional tasks. Pilots for the evaluations were students from the Test Pilot School. The particular maneuvers are described below.

Precision Tracking (Phases I, II, and III). Air-to-air tracking was commenced with the NT-33A approximately 2,000 feet behind the target aircraft. The pilot trimmed the aircraft prior to start and did not re-trim during the maneuver. At the start of the test, the pilot achieved the aim point as rapidly and aggressively as possible and persistently drove the gunsight pipper to the precise aim point. The aim point was the center of the target fuselage at the wing/fuselage junction. Rudder pedals were not used during the maneuver. The specific tracking task for each configuration consisted of the following:

1. Two 280 KIAS 2-g turns in opposite directions for a heading change of approximately 180 degrees.

2. Two windup turns in opposite directions maintaining 280 KIAS from one to 3.5 g's at an onset rate of 0.1 g/second.

In each maneuver the pilot's task was to continuously minimize the error between the aimpoint and the pipper position.

Formation (Phase II). The target aircraft performed a series of lazy-8 maneuvers with up to 90 degrees of bank with the NT-33A in close formation. A windup turn to 3.5 g's was also included.

Gross Maneuvers - Acquisition (Phases I and II). Initial gross acquisition of the target at the start of the precision tracking maneuver was evaluated as a task. In addition, turn reversals were accomplished following each windup turn. The NT-33A was positioned 1500 feet behind the target aircraft. The target then established a 3-g turn at 280 KIAS. Upon call from the NT-33A, the target executed a rapid, unloaded reversal. The evaluation pilot delayed 3 to 4 seconds, then reversed and attempted to rapidly reacquire the target. This task, which required large, sustained inputs, was evaluated on the ability to reacquire the target.

Gross Maneuvers - Cine-Track (Phase II). The maneuver began with the target aircraft in a 2.5-g turn and the NT-33A positioned 1500 feet behind. Upon call from the NT-33A, the target performed a 2.5-g barrel roll through 540 degrees of bank. The maneuver was then repeated in the opposite direction.

Landing (Phases I, II, and III). Configurations were examined during normal touch-and-go landings. In Phases II and III the landings were repeated with an intentional displacement from the glide slope and the center-line to evaluate ease of recovery.

Test Techniques. The evaluation pilots were not informed as to which configurations they were evaluating. The maneuvers were flown as often as the pilot felt necessary to evaluate the configuration. The gun camera film was run during each task, and pipper position error was later analyzed to help in assessing performance. Following each maneuver, the pilot recorded his comments on tape and assigned a rating from the Cooper-Harper rating scale shown in figure 5 (reference 9). Post flight debriefings were conducted.

EVALUATION PILOTS

The evaluation pilots all had a considerable number of flying hours. Their experience varied from a low of 1100 hours to a high of 3000 hours and from B-52 aircraft to air combat in F-4 or Mirage aircraft. Their experience is tabulated below:

NAME	RANK	PHASE	TOTAL FLYING TIME	AIRCRAFT EXPERIENCE
Cima	Lt(USN)	I	1100 Hrs	F-4J, RF-4C, T-38A
LeBeau	Capt	I	1500 Hrs	B-52G, T-38A, RF-4C
Stebe	Capt	I	3000 Hrs	T-38A, U-2, T-33, B-66
Saxon	Capt	II	1850 Hrs	F-4C/D/E, F-100D/F
Haas	Capt	II	2700 Hrs	T-38A, C-130A/E, T-39
Daniel	Capt	II	2100 Hrs	T-38A, O-2A, A-1E/H, T-39
LeBarge	Capt	II	2200 Hrs	KC-135A, C-123K, T-39
Lewis	Capt	III	1150 Hrs	F-4C/D/E, T-38A
Tilden	Capt	III	2300 Hrs	F-4C/D/E, T-38A
Shmul	Capt(Israel)	III	2900 Hrs	Fuga-Magister, Mirage

SUMMARY OF RESULTS BY PHASE

Phase I. Four levels of stick force/response ratio and displacement/force ratio were investigated. Forces ranged from "heavy" to "very light", and displacements varied from "very small" (near fixed stick) to a fairly large amount of stick motion. The evaluation maneuvers consisted primarily of precision tracking. Air-to-ground and landing maneuvers were also evaluated, but little data were obtained and the results are not discussed.

The results of the air-to-air tracking investigation are summarized in figures 6 and 7 in terms of individual pilot ratings with a synopsis of comments for each configuration. In general, the pilots prefered the larger control force gradients (configurations 13, 14, and 15 in figures 6 and 7) and to a lesser degree, smaller control stick motion with heavier control force gradients (configurations 4 and 7). The former set of configurations fell on the edge of the test matrix; thus the extent of this area of preference could not be determined. the pilots indicated that, within this area, control motion was noticeably large but not uncomfortable. The configurations rated the poorest were

the lighter force gradients and the smaller control motions (configurations 1, 2, 3, and 5) and the heavy force gradient with the larger motion (configuration 16). The lighter force/smaller motion configurations resulted in longitudinal and lateral sensitivity. The remaining configurations (6, 8, and 9 through 12) fell in the mid-range of the control motions investigated. They were rated alike - all had average ratings of 4 1/2; however, the pilot comments showed a trend from oversensitivity to sluggishness as forces increased.

A comparison of results of this and of the Calspan test showed similar trends with the exception that ratings and comments for this phase indicated a preference for the larger stick deflections in the very light to medium force gradient range; a result not evidenced in the earlier data.

Phase II. Additional stick force and displacement configurations were investigated in this phase. The evaluation maneuvers consisted of formation, precision tracking, gross acquisition, "Cine-Track", and landing.

Each air-to-air maneuver was rated and commented upon separately. An overall rating was also assigned to the air-to-air tasks. The overall ratings are depicted in figure 8 with pertinent pilot comments summarized in figure 9.

When presented with very heavy force gradients, the pilots found it difficult to make fine corrections during tracking tasks if trimming was avoided during the maneuver. Given the mechanical configuration of the sidestick used in this evaluation, the pilots complained of excessive wrist bending when approximately 20 degrees of deflection was reached.As this coincided with full throw, encountering the stick stop was an additional concern. The loss of stick motion while force command was still available resulted in an inability to make precise corrections. When force or deflection was increased further, the pilots described the aircraft as sluggish. There was a tendency for the aircraft to wander off target and gross acquisition was difficult. With low force gradients, especially when combined with small motion, the aircraft was described as too sensitive and pilot induced oscillation (PIO) tendemcies were apparent. Utilizing the results obtained from this evaluation, an additional configuration was chosen that fell within the region of best performance. Additional flights confirmed the optimum location (figure 9). Configurations using large displacements with light force gradients were not evaluated in this phase.

Phase III. The influence of control system pre-filters was investigated for two force deflection configurations evaluated earlier. These were the "optimum" of Phase II and one with the same force/response ratio but a smaller amplitude of stick motion.

Initially, five pre-filter constants were chosen for each axis with values of 2, 4, 8, 12, and 16 radians/second. Phases I and II employed pre-filters with corner frequencies of 16 and 4 radians/second for air-to-air tracking and for landing, respectively. Following a brief, four flight evaluation, the pre-filters for the bulk of the evaluation

were narrowed down to 2, 8, and 16 radians/second. Tasks during this phase included precision tracking, gross acquisition, lazy-8, and landing.

The results are presented in terms of pilot ratings and comments for precision tracking and landing (figure 10). In addition to Cooper-Harper ratings, the pilots assigned a numerical value to each configuration which was based upon an order of preference. Each pilot ranked the configurations he flew on a particular mission in accordance with a preference rating from zero (most preferred) to 10 (least preferred). An overall preference rating was determined for each configuration by determining its rank order for the entire test. Configurations, when rated by the Cooper-Harper method, were rated independent of one another and the ratings varied over a wide range. However, when they were rated in a preferential order, a more consistent result was obtained. For fine tracking, the 2 radian/second pre-filter produced poor results. Pilots complained of sluggish response and high workload with the optimum configuration. With the small deflection configuration, no such clear distinction was evidenced. The landing task showed a definite pilot preference for either the 8 or the 16 radian/second pre-filter. Overall, there was no clear, consistent influence of pre-filter frequency on the acceptability of the two configurations in the air-to-air phase, and it appeared that no selection of pre-filter would have substantially improved one configuration over another.

OVERALL RESULTS

Significant trends can be seen when the results of all the phases are analyzed together. In particular, there were definite areas in the force/response vs deflection/force matrix where performance was significantly better than elsewhere. Conversely, there were other areas where performance was worse than elsewhere. These areas are depicted in figure 11 together with a summary presentation of the averages of all Cooper-Harper ratings for each configuration. The area of best performance is denoted as "adequate". From a compilation of comments, this term fairly describes the pilot acceptance of these configurations. The worst areas are defined simply as "poor". They do not solely encompas the unacceptable configurations, but rather, represent a region where performance of the task requires noticeably more compensation. It also denotes the regions of the matrix where unacceptable performance may be encountered. Between these regions, the results were either "less than adequate" or not well enough defined by the data to base any conclusion.

Certain conclusions drawn in refrence 3 are substantiated by the current results. In the region of very small motion (fixed stick in refrence 3), tracking performance was very sensitive to changes in force/response gradient. There appears to be a narrow area in the medium to heavy force range where performance in air-to-air maneuvers was adequate. As motion increased, the region expands to a point where the results were fairly insensative to changes in force/response gradients.

As stick motion was increased further, comments about excessive stick motion, encountering the stops, and overshoots arose. The region of very light to medium forces with small to moderate stick motion resulted in very sensitive response and a degradation in performance. From pilot comments, it appeared that a fixed stick or a small displacement stick would not give adequate performance in this range.

A comparison of results for both gross maneuvering and precision tracking indicated no noticeable disparity in the results obtained from the two types of maneuvers. The boundaries of figure 11 apply to all air-to-air maneuvers investigated.

The pilots in all phases complained of a lack of good control force harmony for a significant number of configurations. This problem did not appear to be consistent, and complaints relating to control harmony varied with the configuration. It was obvious, however, that harmony was not optimum for a very large number of configurations.

The tracking technique used in the investigation was intended to identify deficiencies in the longitudinal and lateral axis, and did not allow the use of rudder. The comments of objectionable lateral performance may have been accentuated by this technique. However, when rudder was used, as in the Cine-Track maneuver, the lack of harmony between the rudder and the other controls was apparent, and tracking performance was not improved.

Even in the area of "adequate" performance, very few ratings better than four were obtained. It is felt that the ratings across the matrix were uniformly degraded because of the combination of lateral deficiencies and the tracking technique.

CONCLUSIONS

The following conclusions are applicable to the NT-33A in the particular control system and aircraft configuration evaluated. However, they may be extended in general to other configurations, fighter aircraft, or maneuvers although verification is required. The conclusions pertain to the air-to-air phase only.

1. For very small displacements (including fixed stick), the range of stick force/response gradients that result in adequate performance is very restricted. Lighter or heavier forces will result in objectionable or unacceptable handling qualities.

2. There is a region of moderate stick displacements where performance is relatively insensitive to variations in force/response gradients.

3. Moderate stick motion coupled with light to moderate stick force gradients results in the best tracking performance.

4. The effect of control system prefilters was not significant.

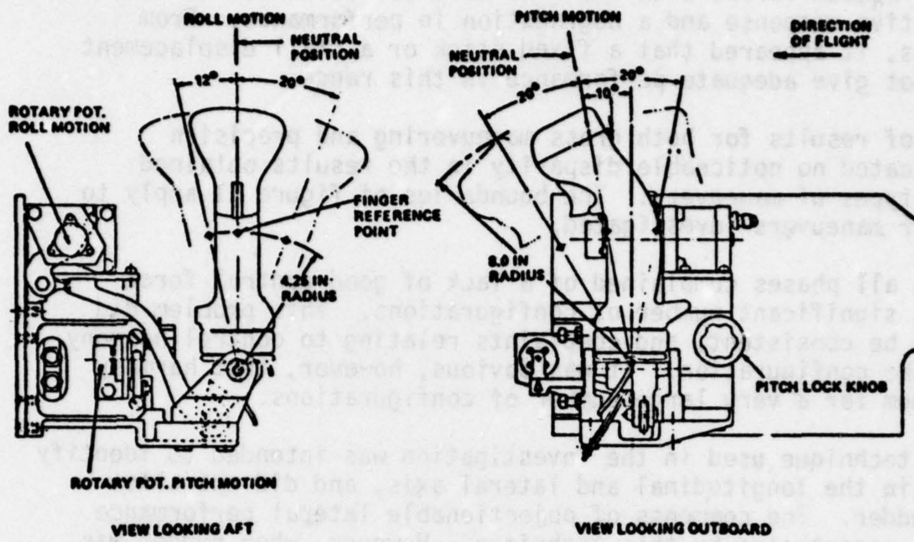

FIGURE 1 NT-33A SIDESTICK

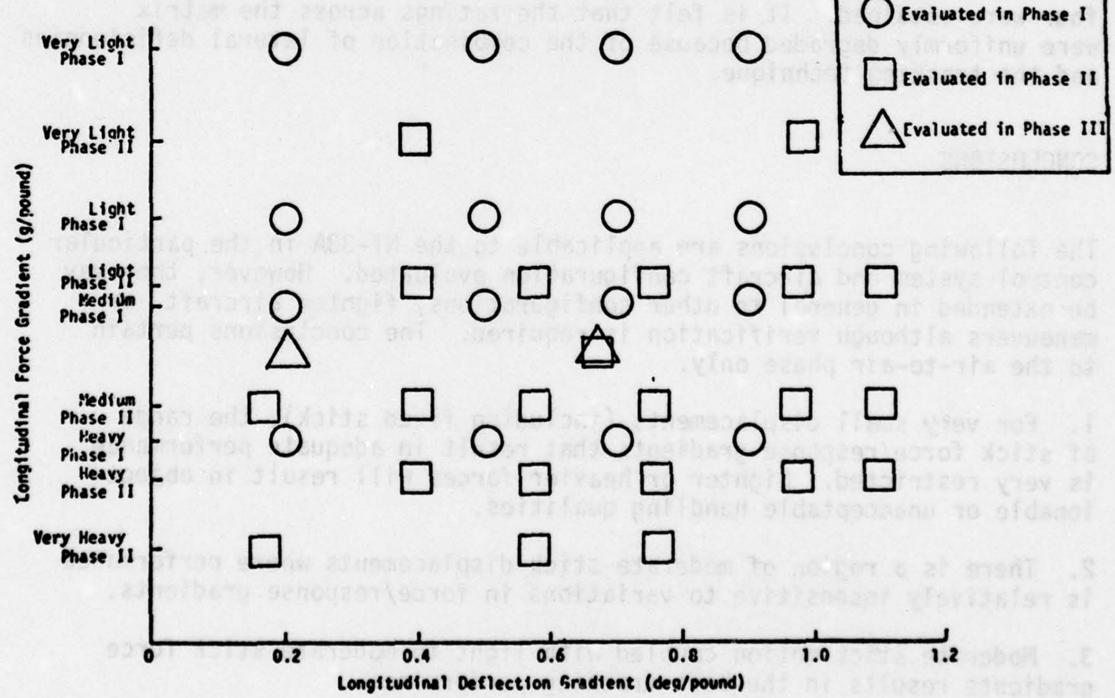

FIGURE 2 MATRIX OF TEST CONDITIONS

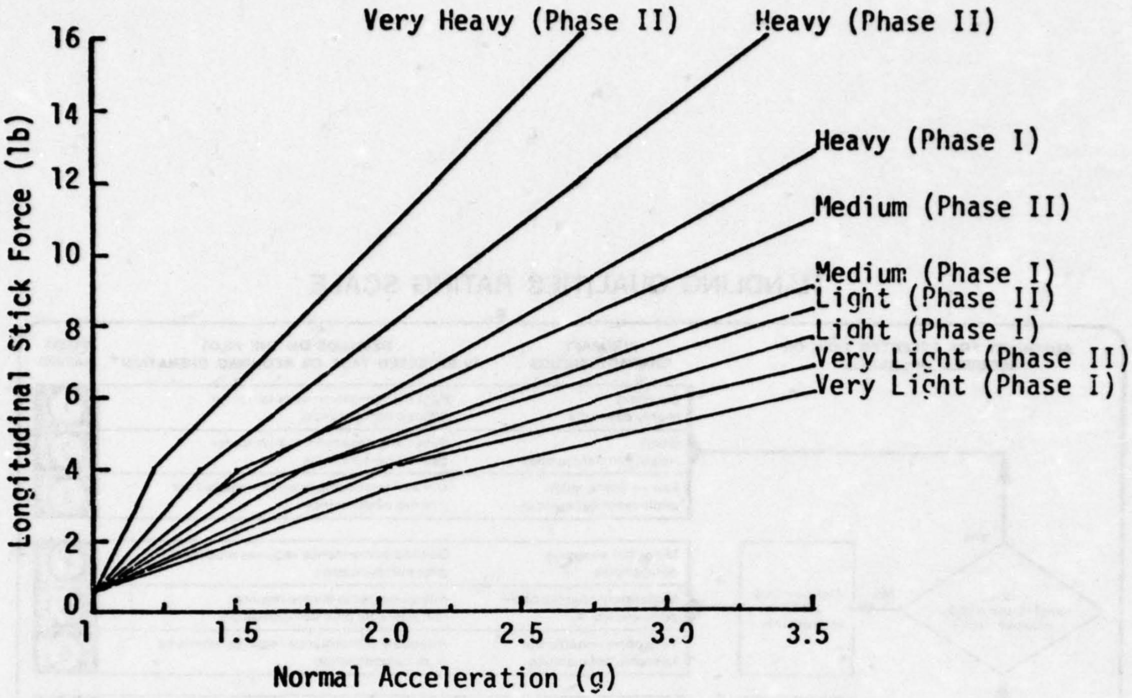

FIGURE 3 LONGITUDINAL STICK FORCE VS NORMAL ACCELERATION

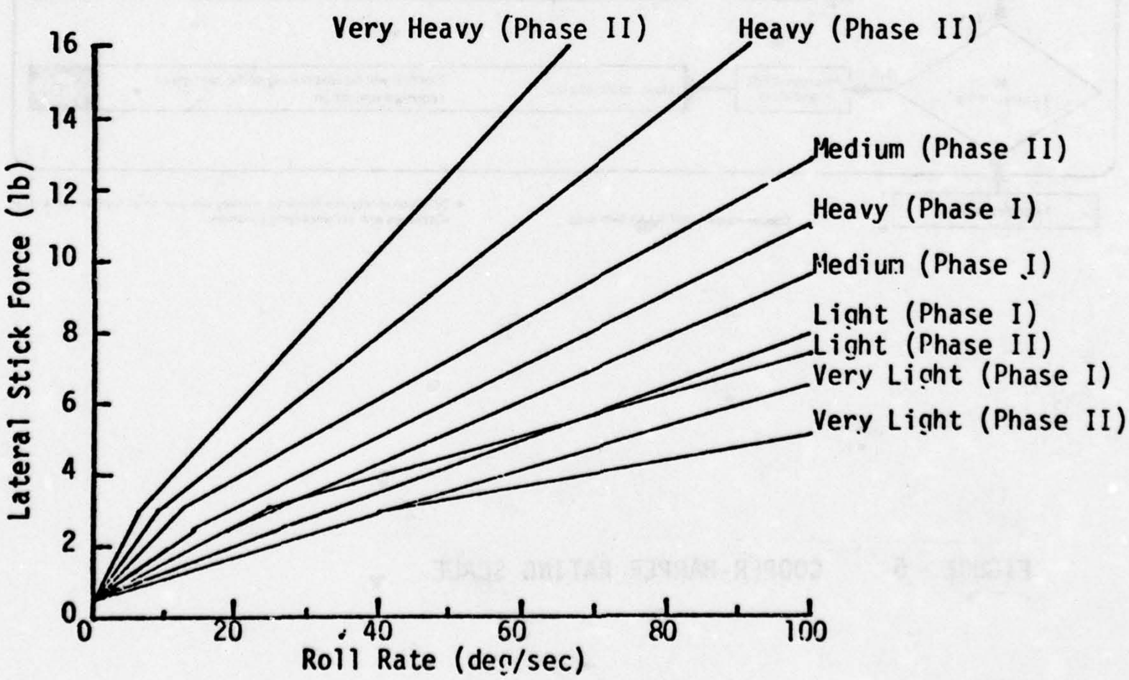

FIGURE 4 LATERAL STICK FORCE VS ROLL RATE

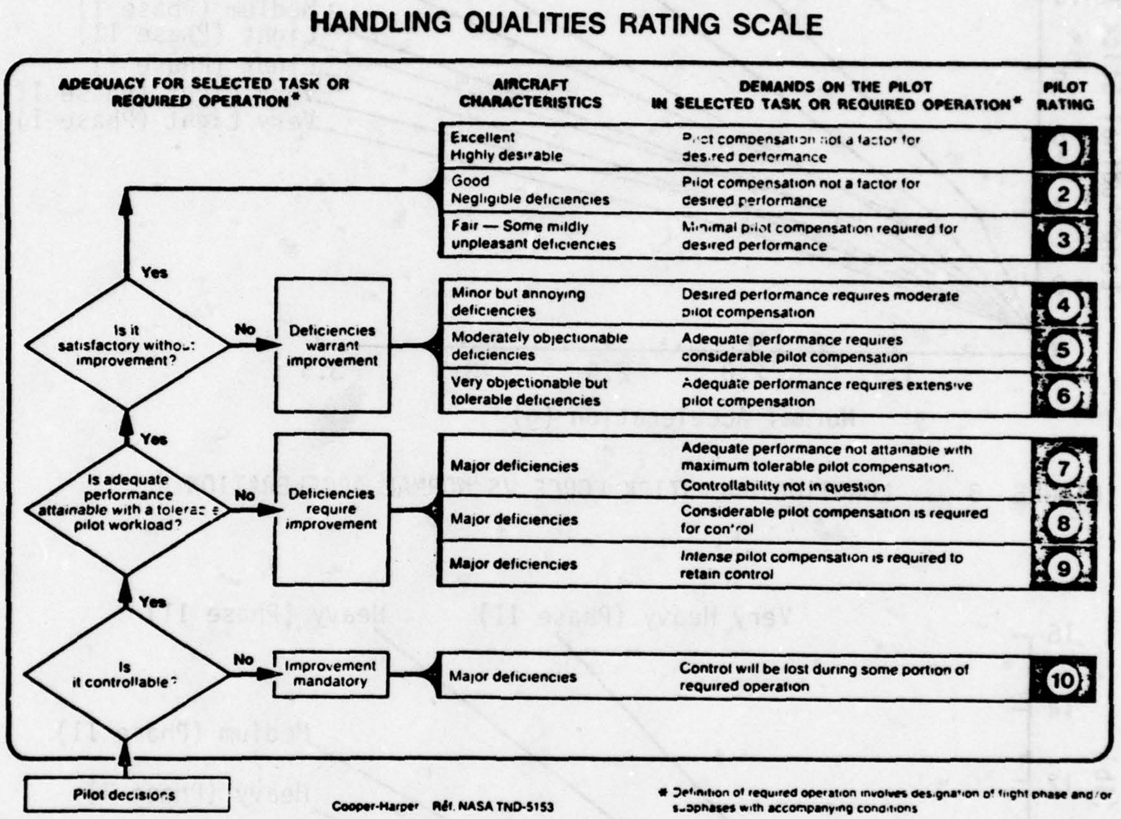

FIGURE 5 COOPER-HARPER RATING SCALE

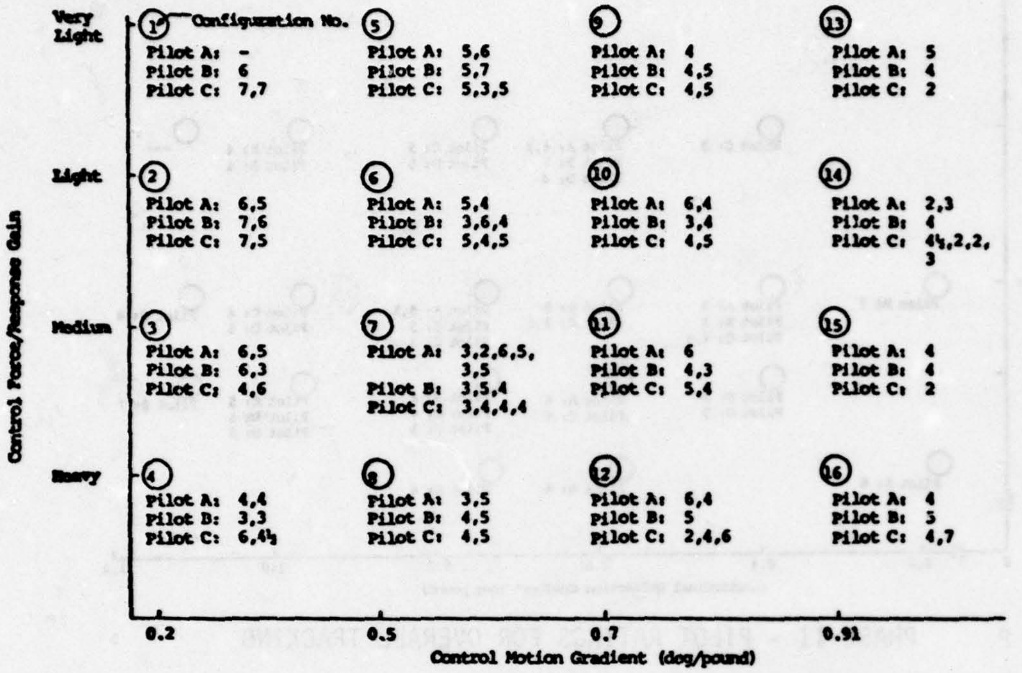

FIGURE 6 PHASE I - PILOT RATINGS FOR AIR- TO-AIR TRACKING

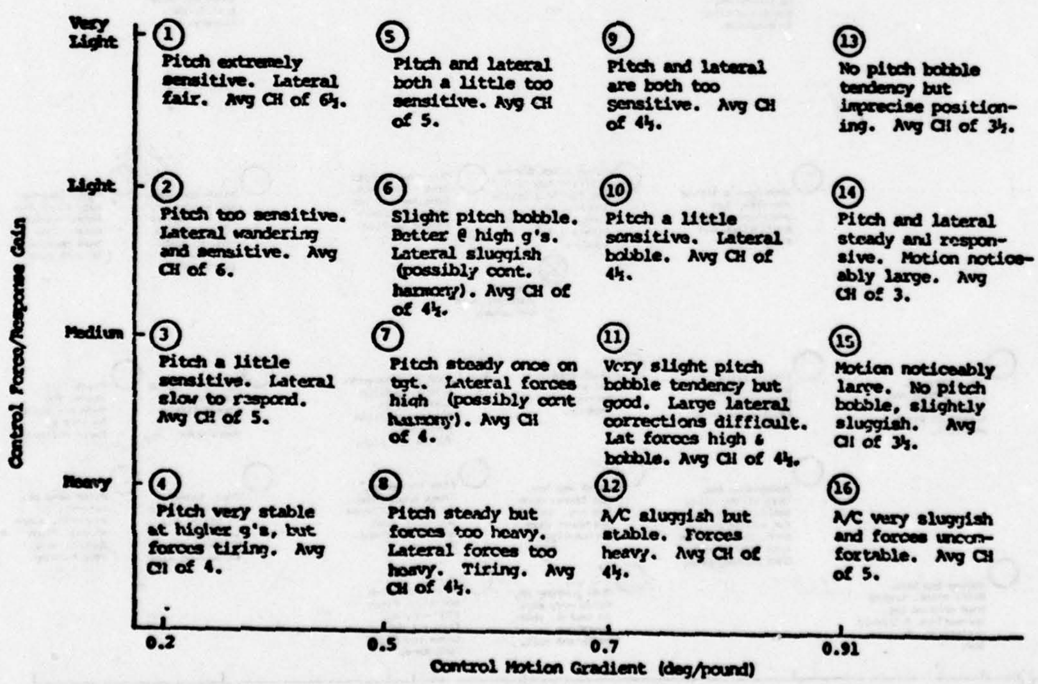

FIGURE 7 PHASE I - PILOT COMMENTS FOR AIR- TO-AIR TRACKING

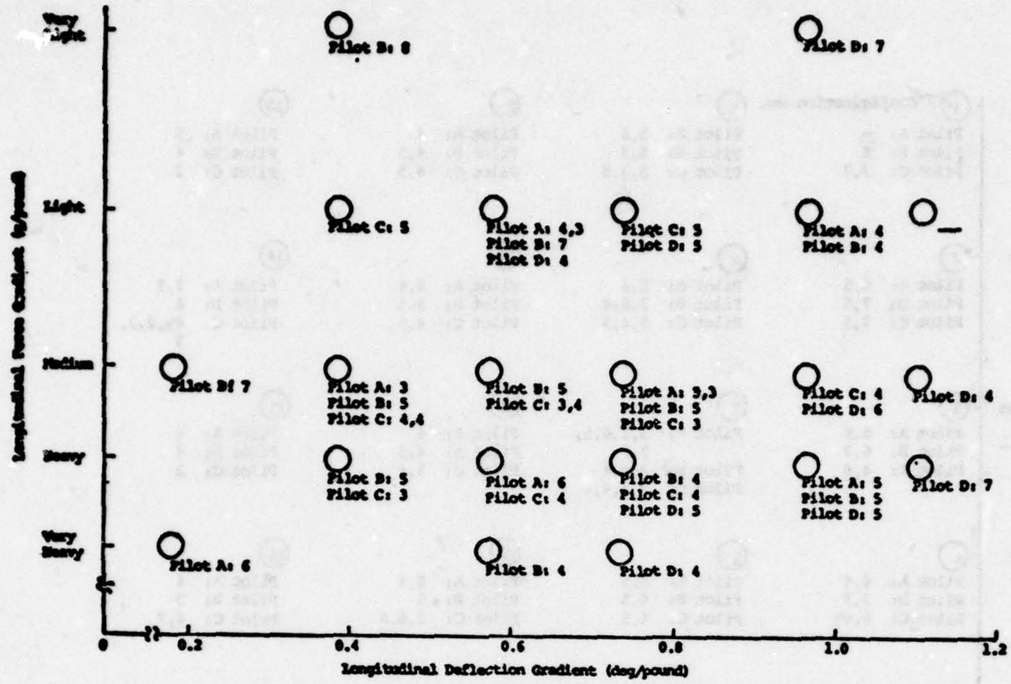

FIGURE 8 PHASE II - PILOT RATINGS FOR OVERALL TRACKING

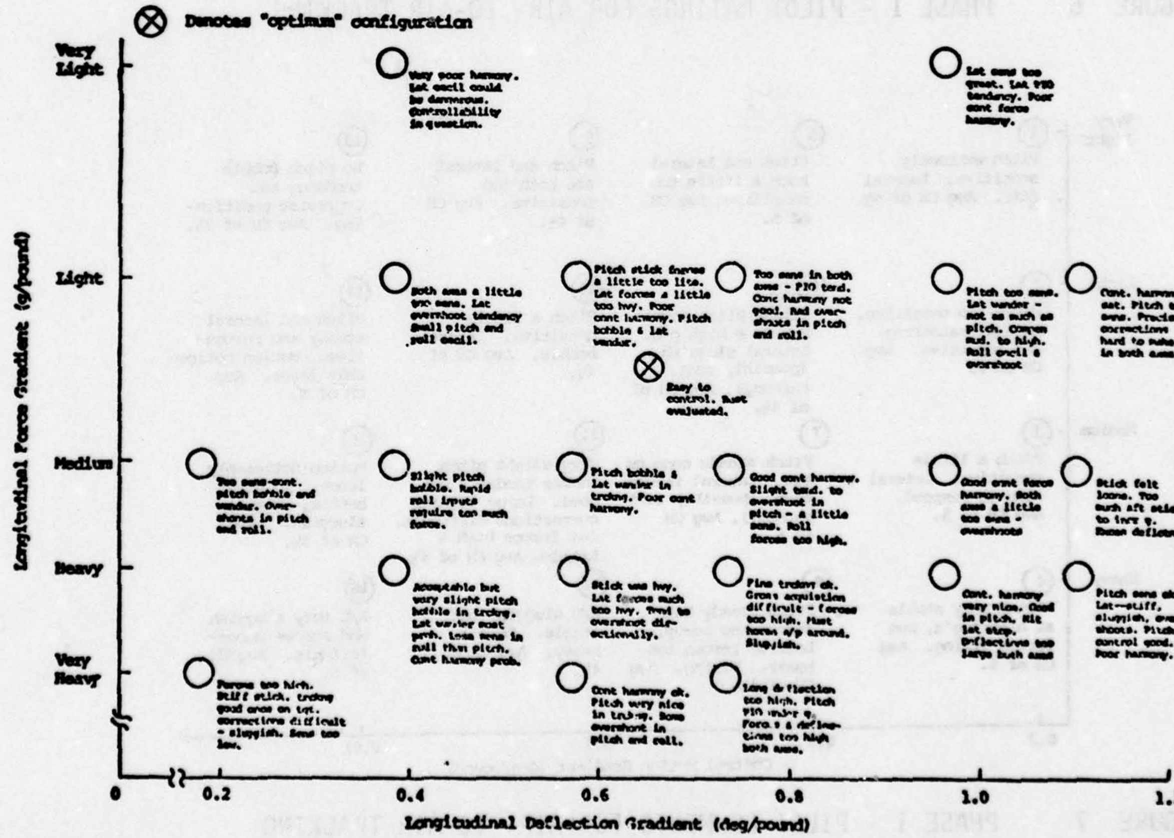

FIGURE 9 PHASE II - PILOT COMMENTS FOR OVERALL TRACKING

Prefilter Corner Frequency (radians/second)	Configuration	
	Optimum Def gradient: 0.7 deg/lb	Small Deflection Def gradient: 0.2 deg/lb
2	Pilot Comments: Continual overshoots. Not responsive. Unpredictable in all tasks. Cooper-Harper Rating 4-8 Preference Rating 8	Pilot Comments: Continual overshoots. Sensitive in g, PIO tendency. Unpredictable in all tasks. Cooper-Harper Rating 4-9 Preference Rating 7.5
8	Pilot Comments: A bit sluggish. Tendency to over control. Noticeable overshoots. Tendency to balloon in landing. Cooper-Harper Rating 4-7 Preference Rating 5	Pilot Comments: Continual overshoots. Too sensitive. Poor predictability. Quite a bit of bobble. Tendency to balloon in landing. Cooper-Harper Rating 2-8 Preference Rating 7
16	Pilot Comments: Sensitive but easy to control. Not g sensitive. Poor predictability but liked the results over-all. Easy to control in landings. Cooper-Harper Rating 3-7 Preference Rating 4.5	Pilot Comments: Poor predictability but can get desired performance. Lateral steps with roll rate. Tendency to over control in roll. Too sensitive in landings. Cooper-Harper Rating 4-8 Preference Rating 7.5

FIGURE 10 PHASE III - PILOT RATINGS AND COMMENTS FOR ALL TASKS

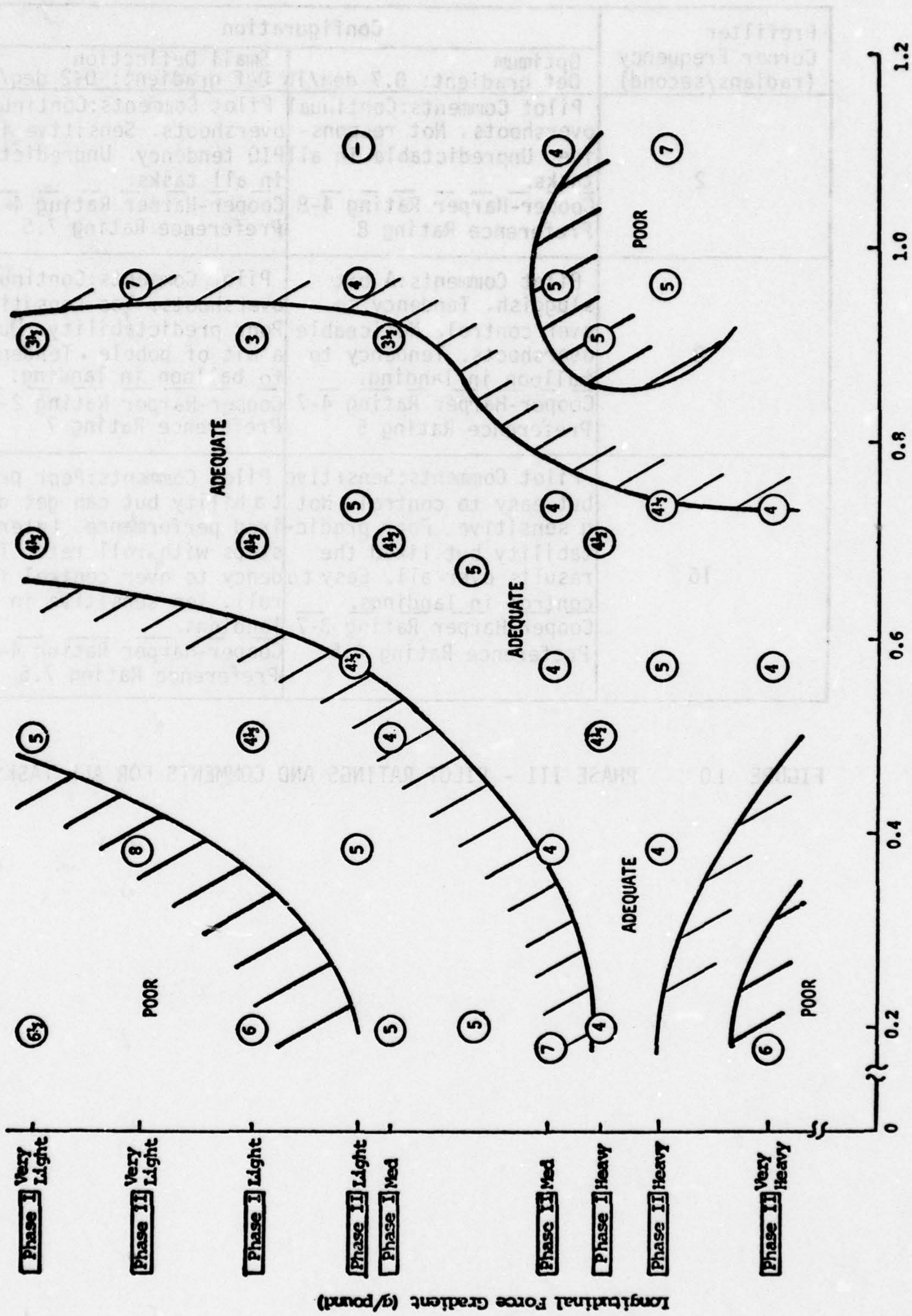

FIGURE 11 AIR-TO-AIR TRACKING PERFORMANCE LEVELS

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8 Tracking Test Techniques for Handling Qualities Evaluation, Twisdale, Franklin, AFFTC-TD-75-1, Air Force Flight Test Center, Edwards Air Force Base, CA, May 1975.

9. The Use of Pilot Rating in the Evaluation of Aircraft Handling Qualities, Cooper, Harper, NASA-TN-D-5153, Ames Research Center, Moffett Field, CA, April 1969.

Rogers Smith, Calspan: What happened to the good configuration of Phase II?

Answer: The "good" is also depicted on the overall summary plot of average Cooper-Harper ratings. It fell within the area defined as "acceptable" performance. It represents the preferred configuration evaluated in Phase II.

Phil Brown, NASA Langley: Can you compare center-stick and side-sticks?

Answer: No evaluations were conducted by the Test Pilot School in this series of tests with a center-stick.

Question: Was there an armrest?

Answer: Yes, it was adjustable vertically and, to a very limited extent, in tilt. There were a few complaints about it but that is a human factors problem and could not be evaluated.

Bud Iles, Grumman: Were there any coupling problems?

Answer: Yes, there was a continual lateral/directional wander throughout all three phases.

Tim Sweeney, ASD: Did you vary breakout forces?

Answer: Breakout forces were fixed at 0.5 pounds for both axes.

Jim Clay: Did you do an analysis of tracking performance?

Gun camera film was analyzed & tracking error time histories were obtained. Little correlation was found between quantitative tracking performance and pilot ratings/comments. Apparently the pilots rated how hard they worked to attain that level of performance.

John Schuler, Boeing: Did you look at characteristics of the pilots?

Answer: Generally, pilot ratings were very closely grouped. There were a few wild points but as long as the pilots performed the tracking in the same manner, they grouped closely.

Rogers Smith, Calspan: Were the pilots equally aggressive?

Answer: To our knowledge they were. If you are referring to Phase III, all pilots had similar backgrounds (they were all fighter pilots). All were briefed to fly the task aggressively.

SIFT PILOT-IN-THE-LOOP HANDLING QUALITIES TEST AND ANALYSIS TECHNIQUES

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INTRODUCTION

System Identification From Tracking (SIFT) is a new flight test and analysis technique for evaluating pilot-in-the-loop handling qualities. Normal stability and control flight test data are obtained during pilot-in-the-loop, mission-oriented, precision tracking maneuvers. These data are analyzed in the frequency domain to obtain frequency response transfer functions and modal parameters of the airplane aerodynamics, the flight control system, and the overall system (aerodynamics plus control system). These quantitative results are correlated with qualitative pilot comments and ratings obtained from the test maneuver. This correlation of quantitative and qualitative results provides insight into the pilot-in-the-loop handling qualities characteristics of the airplane.

Numerous applications of the SIFT techniques at AFFTC over the past two and a half years include a longitudinal short period pilot-induced-oscillation (PIO) investigation, an evaluation which uncovered the existence of a previously unsuspected lateral-directional coupling into the pitch axis, and a fixed-base simulator investigation.

The purpose of this paper is to briefly and informally outline the test and analysis techniques which make up SIFT. Formal preliminary documentation of the SIFT techniques and of AFFTC experience with these techniques is available in Reference 1. Development of these techniques is continuing at AFFTC, using flight test data and fixed-base simulator studies.

DISCUSSION

A schematic outline of the SIFT techniques is presented in Figure 1. For the purpose of discussion, this outline may be roughly divided into "test techniques" and "analysis techniques". Both quantitative and qualitative data are obtained during the test maneuver. Quantitative data are in the form of time histories of normal stability and control parameters recorded during the maneuver. The qualitative data are in the form of pilot ratings and comments. The quantitative data are analyzed in the frequency domain and the results are correlated with the qualitative pilot comment data to obtain insights into the pilot-in-the-loop handling qualities characteristics of the airplane.

Test Techniques

Test data are obtained during pilot-in-the-loop, mission-oriented, precision tracking maneuvers. Three test maneuvers have been successfully used at AFFTC. One of these is a specially developed "tail chase" precision air-to-air tracking maneuver called a Handling Qualities During Tracking (HQDT) maneuver. The other two are precision formation flying and aerial refueling. The latter two maneuvers are essentially standard maneuvers, flown just as they are flown "in the field".

The HQDT test maneuver is a nominally constant angle of attack, constant Mach number, constant altitude, carefully controlled precision air-to-air tracking maneuver. Special piloting techniques are used to assure good frequency content in the pilot's stick and rudder pedal inputs. These special piloting techniques include agressive and persistent attempts to correct even very small tracking errors. Good frequency content and controlled test conditions are important to a good data analysis. The HQDT test maneuver has been formally documented in Reference 2.

Extensive experience at AFFTC has demonstrated the HQDT test maneuver to be an excellent maneuver for eliciting useful pilot comments as well as good quantitative test data.

Analysis Techniques

The quantitative test data (time histories of stability and control parameters acquired during the test maneuver) are analyzed in the frequency domain to obtain open-loop frequency response transfer functions. These transfer functions may be the airframe aerodynamics, or the flight control system, or the overall airplane (aerodynamics plus control system), or some other component or system (e.g., a filter or a servoactuator). One of the advantages of the frequency response curves identified by SIFT analysis techniques is that they are a measure of the actual system as implemented. They are not based on a presumed model of the system.

After the frequency response curves have been identified, they may be curve fitted using a modal analysis program. The modal analysis program identifies the system gain, transport time delay, and first and second order characteristics. This identification is based on a system model provided by the program user.

The frequency response curves and modal parameters are correlated with pilot ratings and comments to provide insights into the handling qualities characteristics of the airplane being tested. This correlation of quantitative and qualitative data obtained from the same test maneuver has provided some interesting results at AFFTC. In one case, a previously unsuspected cross-coupling of lateral-directional dynamics into the pitch axis was discovered using SIFT techniques. This cross-coupling was evident to the pilot in the form of a two to three mil pitch bobble at twice the dutch roll frequency. A complete classical stability and control test program, conducted prior to the SIFT testing, had not uncovered this cross-coupling problem. In another case, the Smith PIO criteria were used with

SIFT techniques to confirm pilot reports of longitudinal short period PIO. In yet another case, the influence of an air-refueling boom on the dynamics of a refueling aircraft was observed. This was done by comparing the frequency response curves of the refueling airplane before hook-up with the frequency response curves after hook-up.

Example Test Results

A fixed base five degree of freedom engineering simulator was used at AFFTC to obtain test data, using SIFT test techniques. Two simulated HQDT test maneuvers were "flown" to obtain the data discussed here. A target was presented on a cathode ray tube (CRT) display for the tracking pilot. The target was programmed to fly a constant load factor, constant Mach number, level turn (3g, .080 Mach, 15000 ft). Seventy-one seconds of data were obtained during the first HQDT test maneuver, and sixty-three seconds of data were obtained during the second maneuver. The pilot flew these two maneuvers with his feet on the floor, i.e., without making any rudder pedal inputs. The pilot's Cooper-Harper rating for the configuration flown was a 3 for the pitch axis, a 3 for the lateral-directional axes, and a 3 overall.

The data obtained during these two maneuvers were analyzed in the frequency domain to obtain frequency response curves of the flight control system, the airplane aerodynamics, and the overall system (aerodynamics plus control system). The frequency response curves of the flight control system transfer function of stabilator deflection to pitch stick force (δ_e/F_s), along with power spectral density and coherence function plots, are presented in Figure 2. The power spectral density plot of pitch stick force shows the good frequency content elicited by the agressive, persistent, precision piloting technique used in SIFT testing.

The aerodynamic frequency response curves of the pitch rate to stabilator deflection transfer function ($q/-\delta_e$), along with power spectral density and coherence function plots, are presented

in Figure 3. The overall system frequency response curves of the pitch rate to pitch stick force transfer function (q/F_s), along with power spectral density and coherence function plots, are presented in Figure 4.

It is evident from Figures 2 and 3 that a better identification was achieved for the flight control system than for the aerodynamics. This is largely because a very simple pitch axis control system was implemented for this test, consisting only of a stick force gradient and a servoactuator. Because the control system required relatively few components to simulate, there were fewer sources of noise and error compared to the more complex aerodynamics portion of the simulation.

Modal analysis of the identified frequency response transfer functions shown in Figures 2, 3, and 4 yielded the results presented in Figures 5, 6, and 7. The modal parameters are shown in the familiar factored transfer function form. Plots of the modal analysis curve fit overlaid on the identified frequency response transfer function curves are also presented. These overlaid plots provide an indication of how well the modal analysis matched the identified frequency response curves. For example, the short period frequency and damping ratio identified from the aerodynamic frequency response curves are $\omega_{n_{sp}} = 5.69$ radians/second and $\zeta_{sp} = 0.81$. These parameters, with the identified numerator time constant ($1/T_{\theta_2} = 2.04$/second) and gain ($K_q = 1.82 \; \frac{\text{degrees/second}}{\text{degree}}$), provide a pretty good match of the aerodynamics.

<u>Equivalent System Identification Using SIFT Techniques (Testing to the New MILSPEC)</u>

One of the proposed changes to the current flying qualities MILSPEC is the implementation of a second order equivalent system criteria for the pitch axis. An equivalent system criteria would

be easy to test to using SIFT techniques. However, there are some questions concerning standardization of analysis procedures which need to be resolved. Two of these questions are: (1) how will the user initially estimate the equivalent system parameters, and; (2) what will be the frequency range of the curve fit.

Three examples illustrate the problems which may arise from these questions. The following equivalent system model was used in each case:

$$\frac{q}{F_s} = K_E e^{-as} \frac{\left(\frac{S}{1/T_E} + 1\right)}{\frac{S^2}{\omega_E^2} + \frac{2\zeta_E}{\omega_E} S + 1}$$

Table 1 shows the results of three equivalent system fits of the q/F_s transfer function presented in Figure 4. Two fits were constrained to the region zero to ten radians/second. The third fit was over the region zero to fifteen radians/second. Table 1 presents the initial user estimates of the equivalent system modal parameters, the final modal parameter estimates produced by the modal analysis program, and the cost function corresponding to each fit. The cost function provides a measure of how well the frequency response curves in Figure 4 were matched. Smaller cost functions indicate a better match. The respective matches are presented in Figures 8, 9, and 10.

It is apparent from Table 1 that it is possible to obtain different equivalent system matches of the same frequency response curves. The equivalent system parameters identified by modal analysis depend on the user's initial estimate of these parameters and on the frequency range being analyzed. This problem of non-uniqueness of fit has been widely recognized and emphasizes the need to develop uniform procedures for obtaining equivalent system parameters.

CONCLUSIONS

AFFTC experience with SIFT pilot-in-the-loop handling qualities testing has been uniformly encouraging. SIFT has been successfully used to identify and evaluate some unusual handling qualities problems as well as to identify traditional handling qualities parameters such as short period frequency and damping ratio. SIFT techniques are readily adaptable to the equivalent second order system criteria proposed for the revised flying qualities MILSPEC. A standardized procedure should be developed for identifying the equivalent system parameters.

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1. Twisdale, Thomas R. and Ashurst, Tice A., Jr., System Identification From Tracking (SIFT), a New Technique For Handling Qualities Test and Evaluation (Initial Report), AFFTC-TR-77-27, Air Force Flight Test Center, Edwards AFB, California, November 1977.

2. Twisdale, Thomas R. and Franklin, David L., Captain, USAF, Tracking Test Techniques for Handling Qualities Evaluation, AFFTC-TD-75-1, Air Force Flight Test Center, Edwards AFB, California, May 1975.

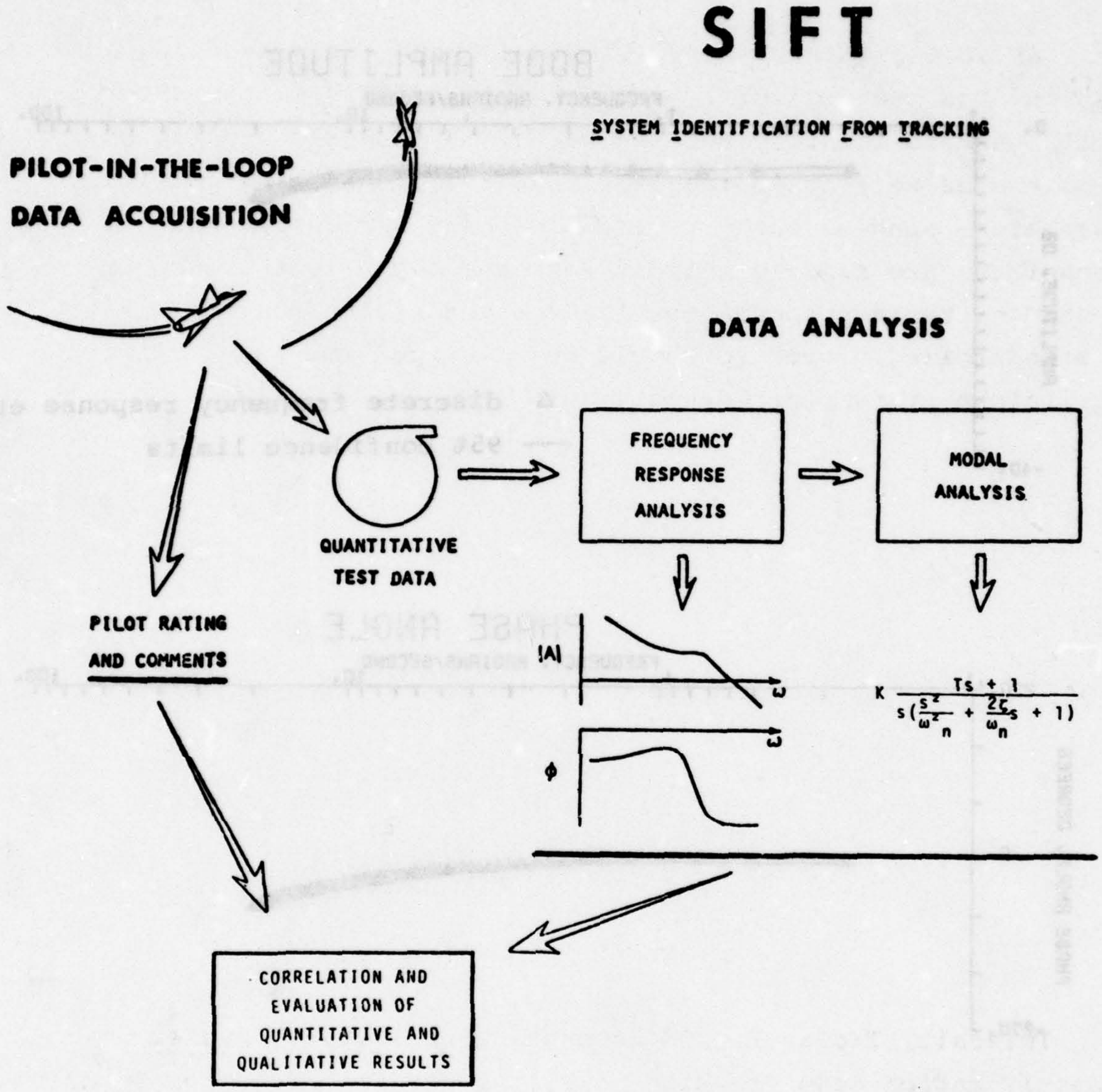


FIGURE 1 SCHEMATIC OUTLINE OF SIFT PILOT-IN-THE-LOOP HANDLING QUALITIES TEST TECHNIQUES

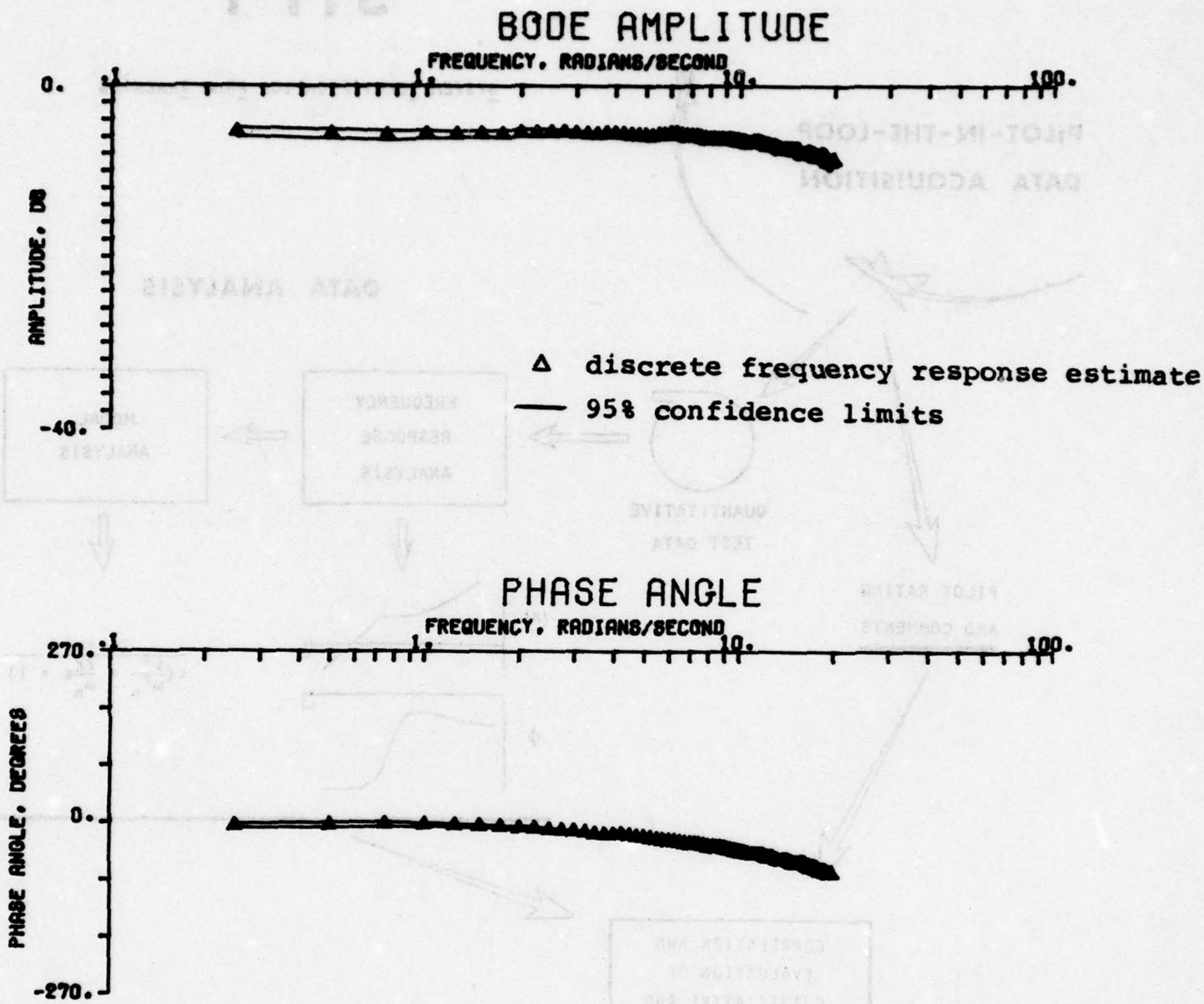


FIGURE 2 FLIGHT CONTROL SYSTEM IDENTIFICATION: FREQUENCY RESPONSE CURVES OF STABILATOR DEFLECTION TO PITCH STICK FORCE TRANSFER FUNCTION $\left(\frac{-\delta_e}{F_s}, \frac{\text{degrees}}{\text{pound}}\right)$

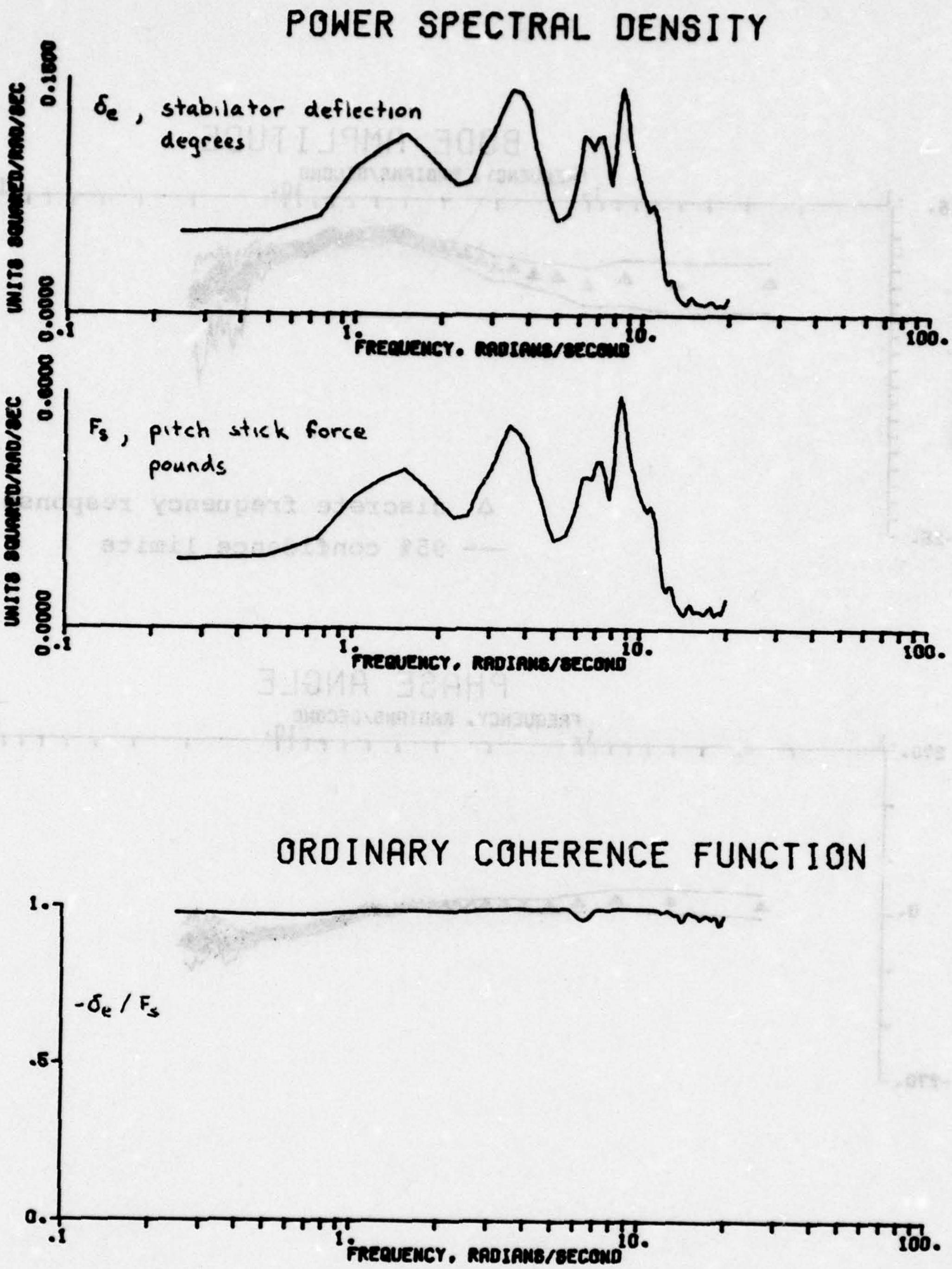
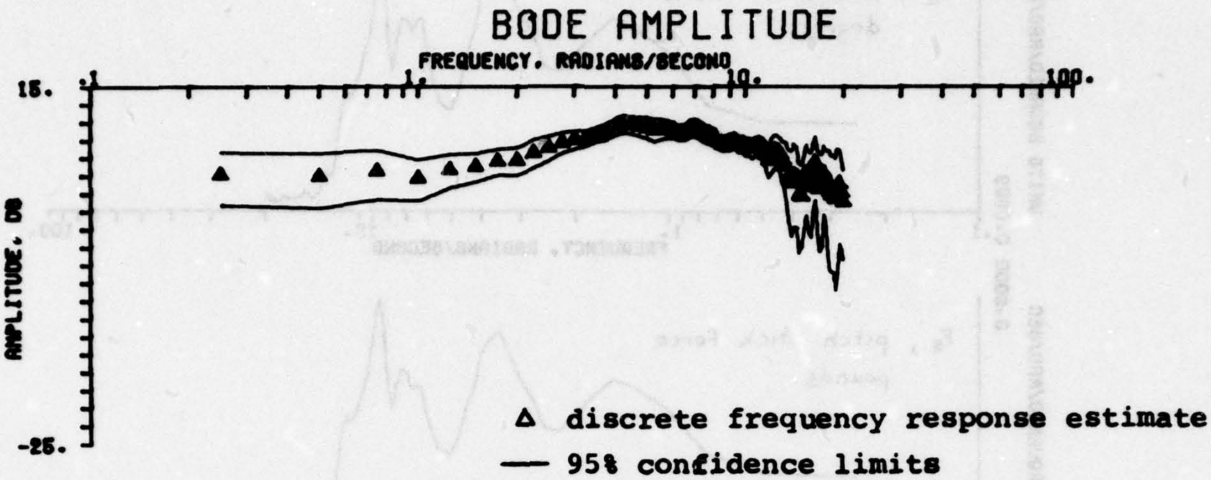


FIGURE 2 (CONCLUDED) **FLIGHT CONTROL SYSTEM IDENTIFICATION: POWER SPECTRAL DENSITY AND COHERENCE FUNCTION PLOTS**



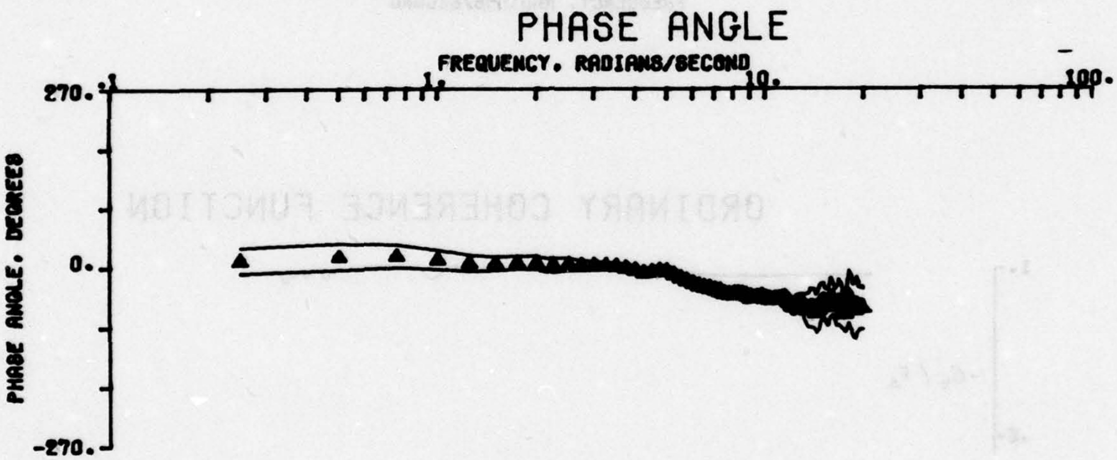


FIGURE 3 AIRFRAME AERODYNAMICS IDENTIFICATION: FREQUENCY RESPONSE CURVES OF PITCH RATE TO STABILATOR DEFLECTION TRANSFER FUNCTION $\left(\frac{q}{-\delta_e}, \frac{\text{degrees/second}}{\text{degree}}\right)$

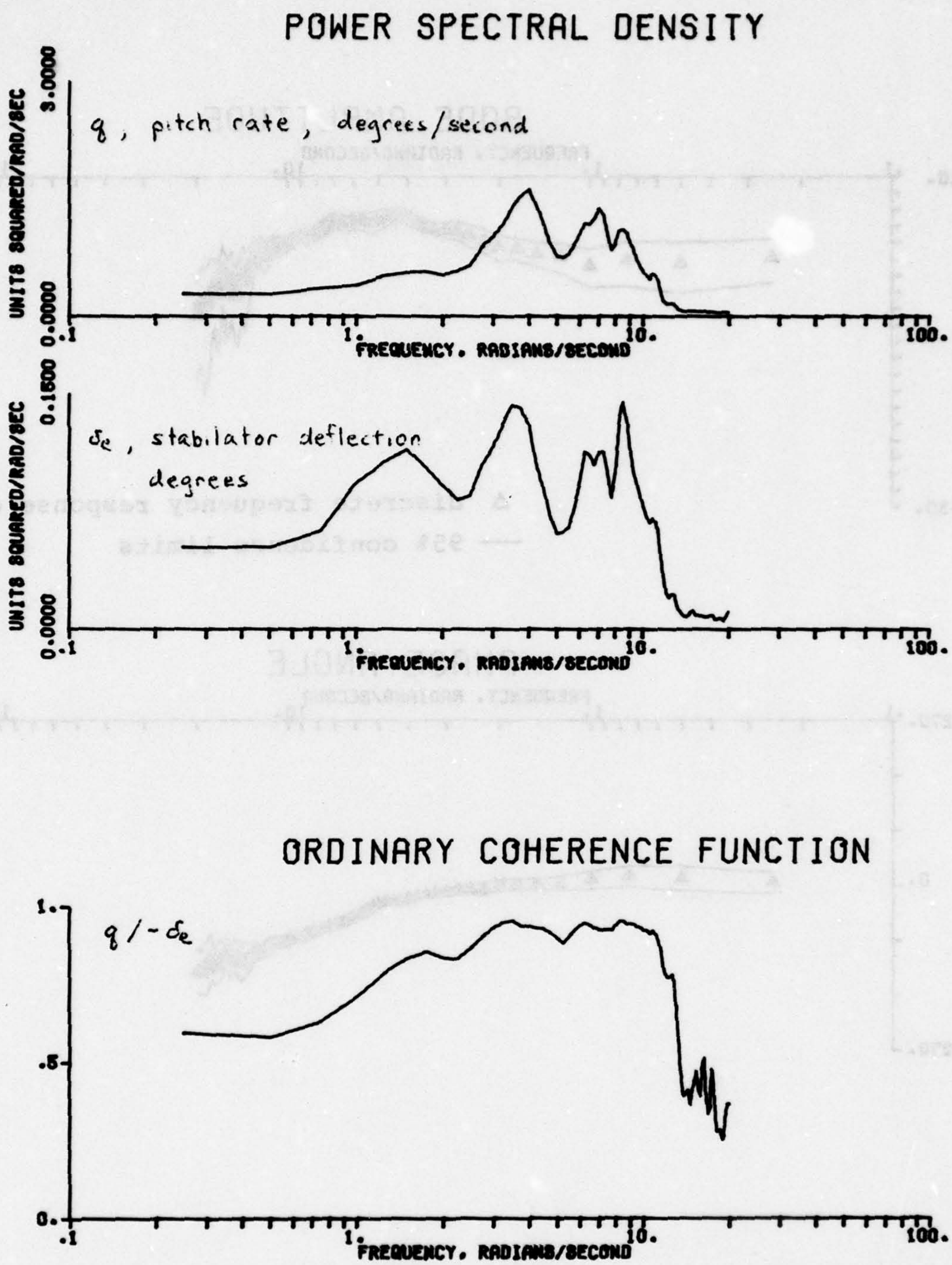
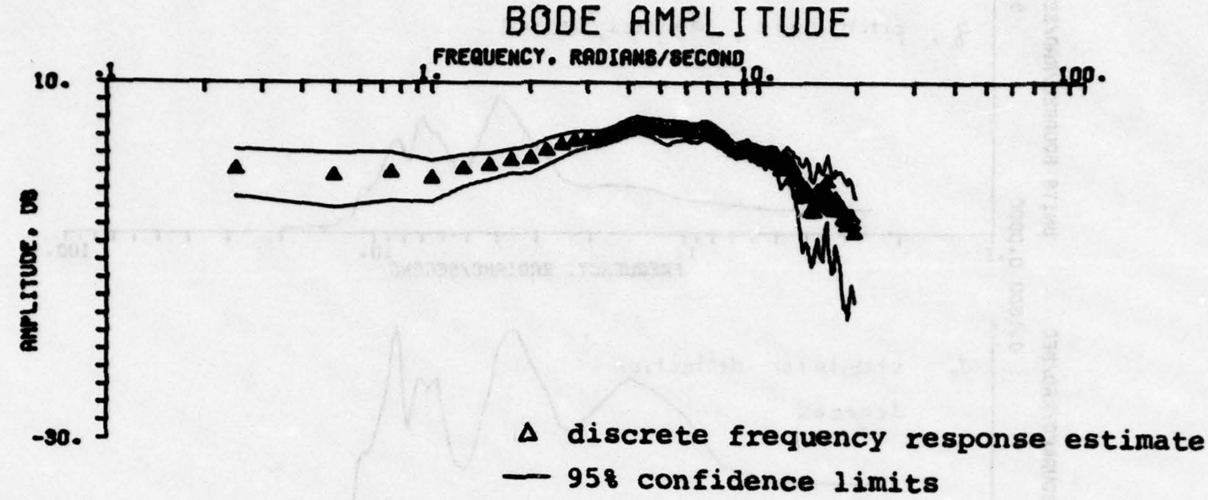


FIGURE 3 (CONCLUDED) AIRFRAME AERODYNAMICS IDENTIFICATION: POWER SPECTRAL DENSITY AND COHERENCE FUNCTION PLOTS



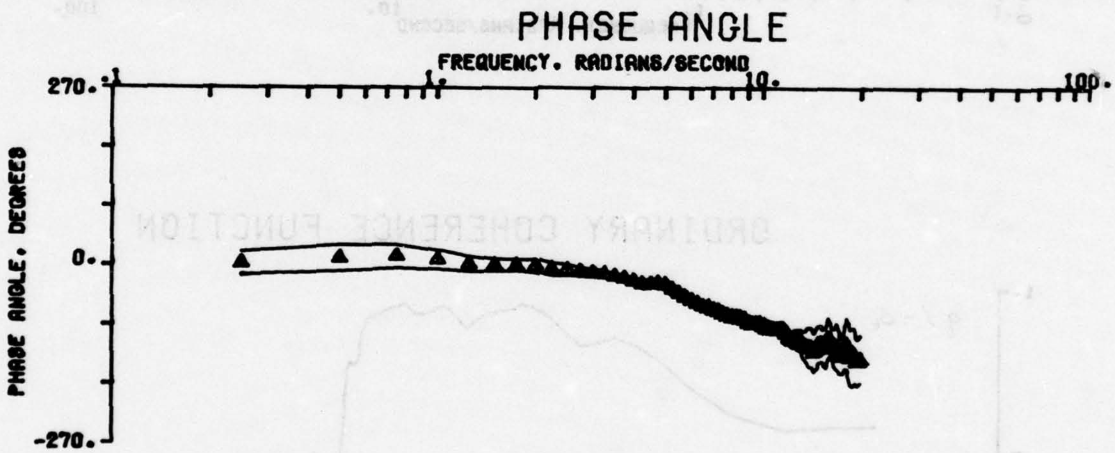


FIGURE 4 OVERALL SYSTEM IDENTIFICATION (AERODYNAMICS PLUS CONTROL SYSTEM): FREQUENCY RESPONSE CURVES OF PITCH RATE TO PITCH STICK FORCE TRANSFER FUNCTION

$$\left(\frac{q}{F_s}, \frac{\text{degrees/second}}{\text{pound}}\right)$$

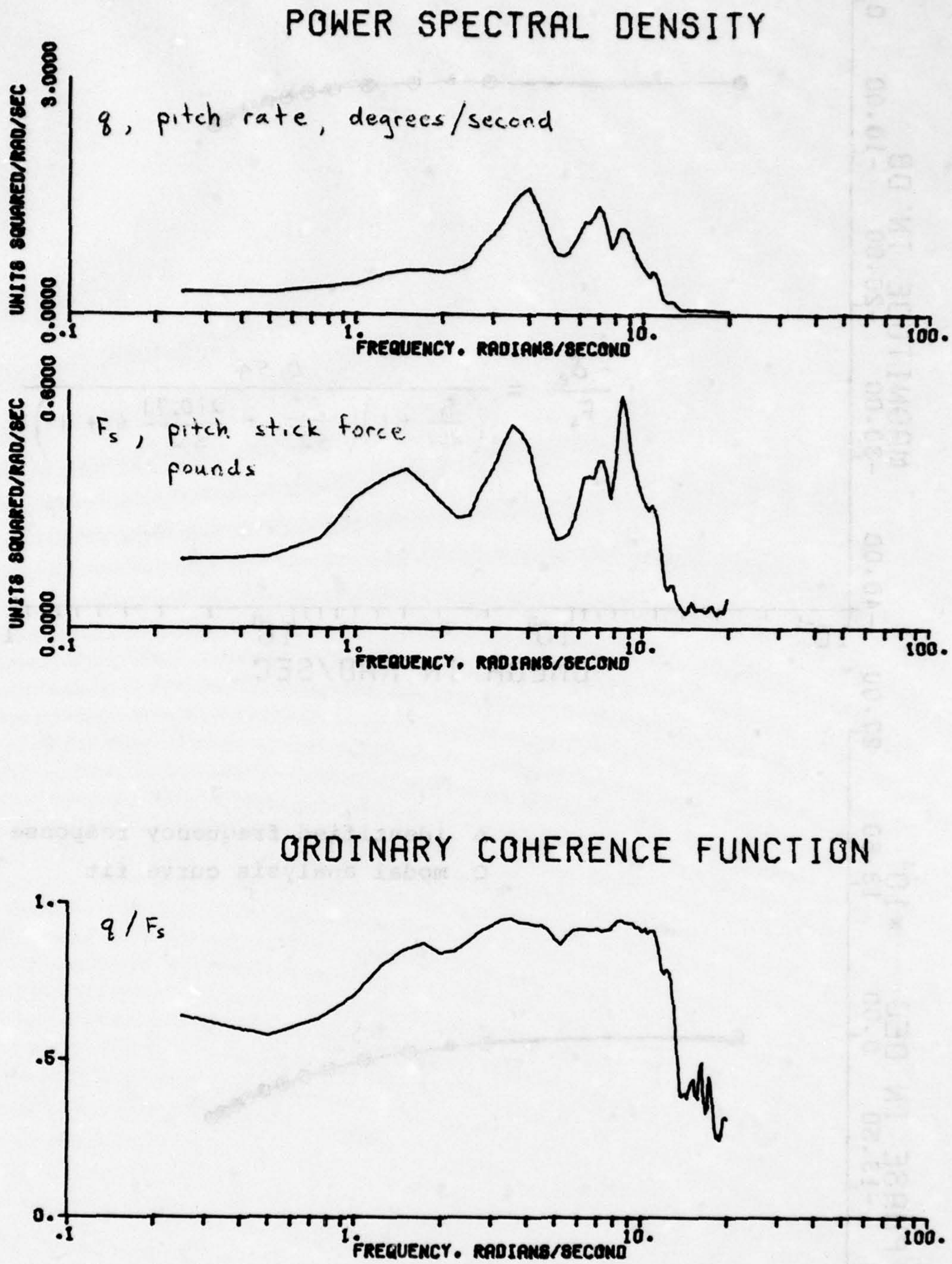


FIGURE 4 (CONCLUDED) OVERALL SYSTEM IDENTIFICATION (AERODYNAMICS PLUS CONTROL SYSTEM): POWER SPECTRAL DENSITY AND COHERENCE FUNCTION PLOTS

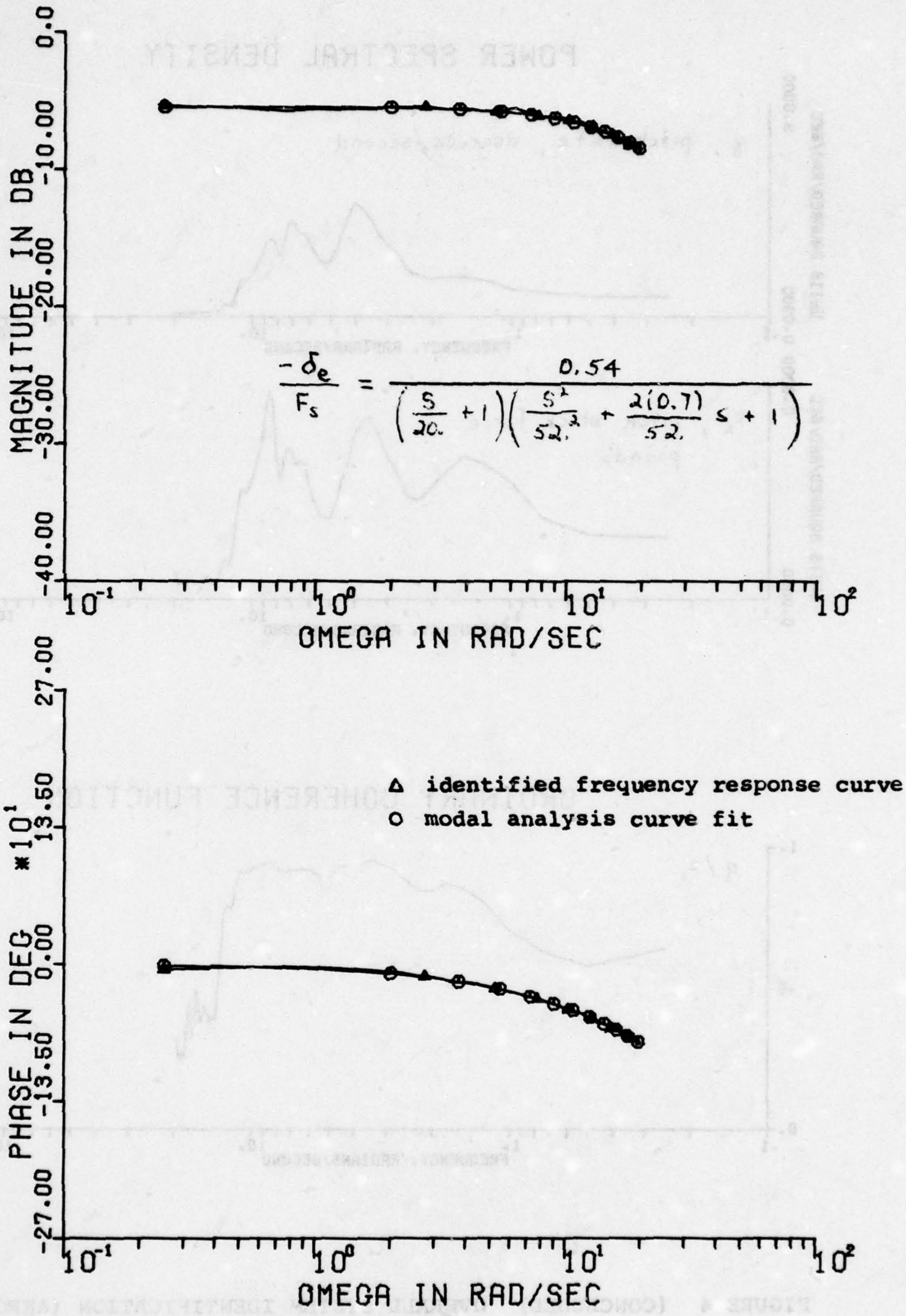


FIGURE 5 RESULTS OF MODAL ANALYSIS OF FLIGHT CONTROL SYSTEM FREQUENCY RESPONSE CURVES

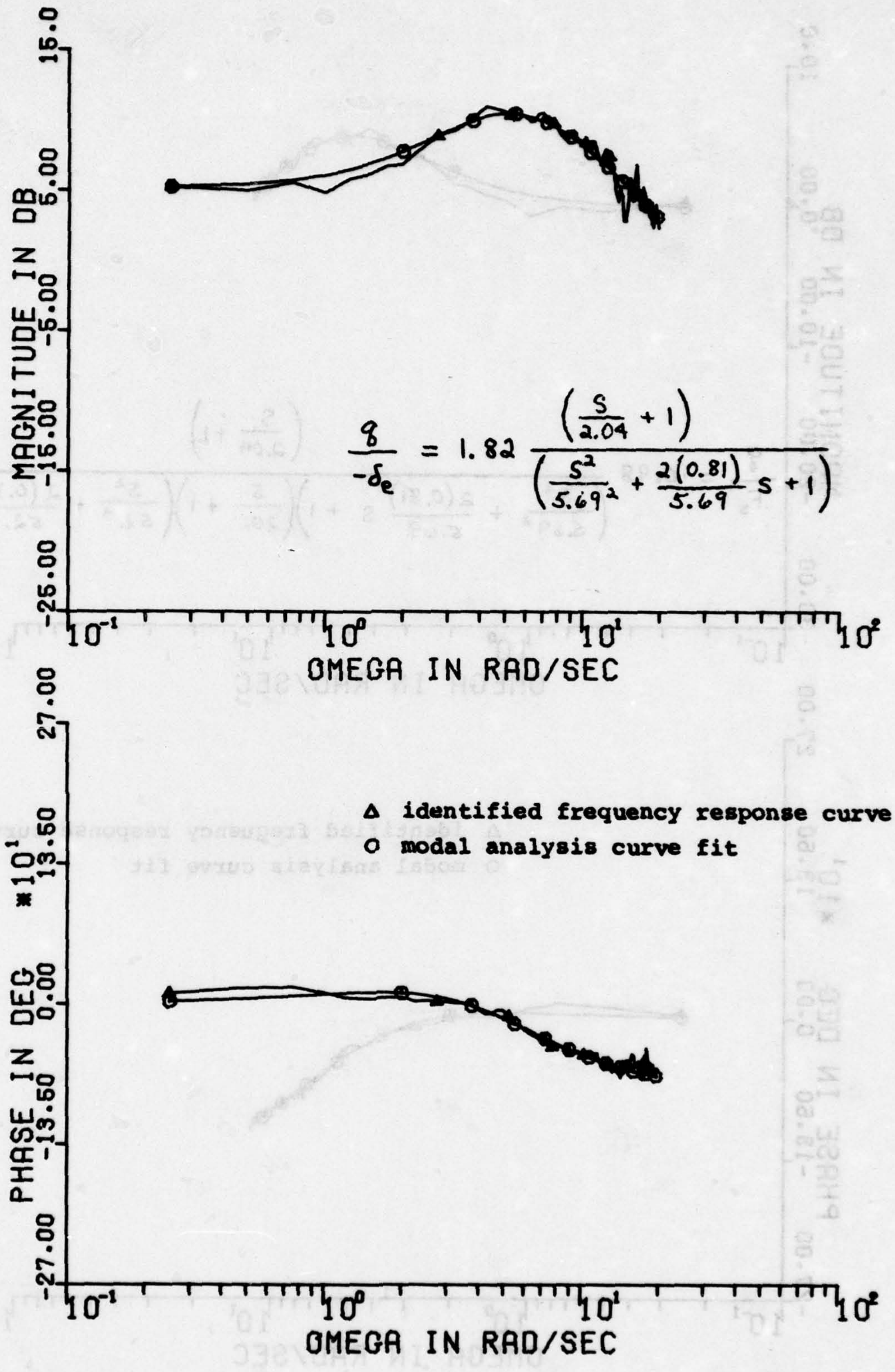


FIGURE 6 RESULTS OF MODAL ANALYSIS OF AIRFRAME AERODYNAMICS FREQUENCY RESPONSE CURVES

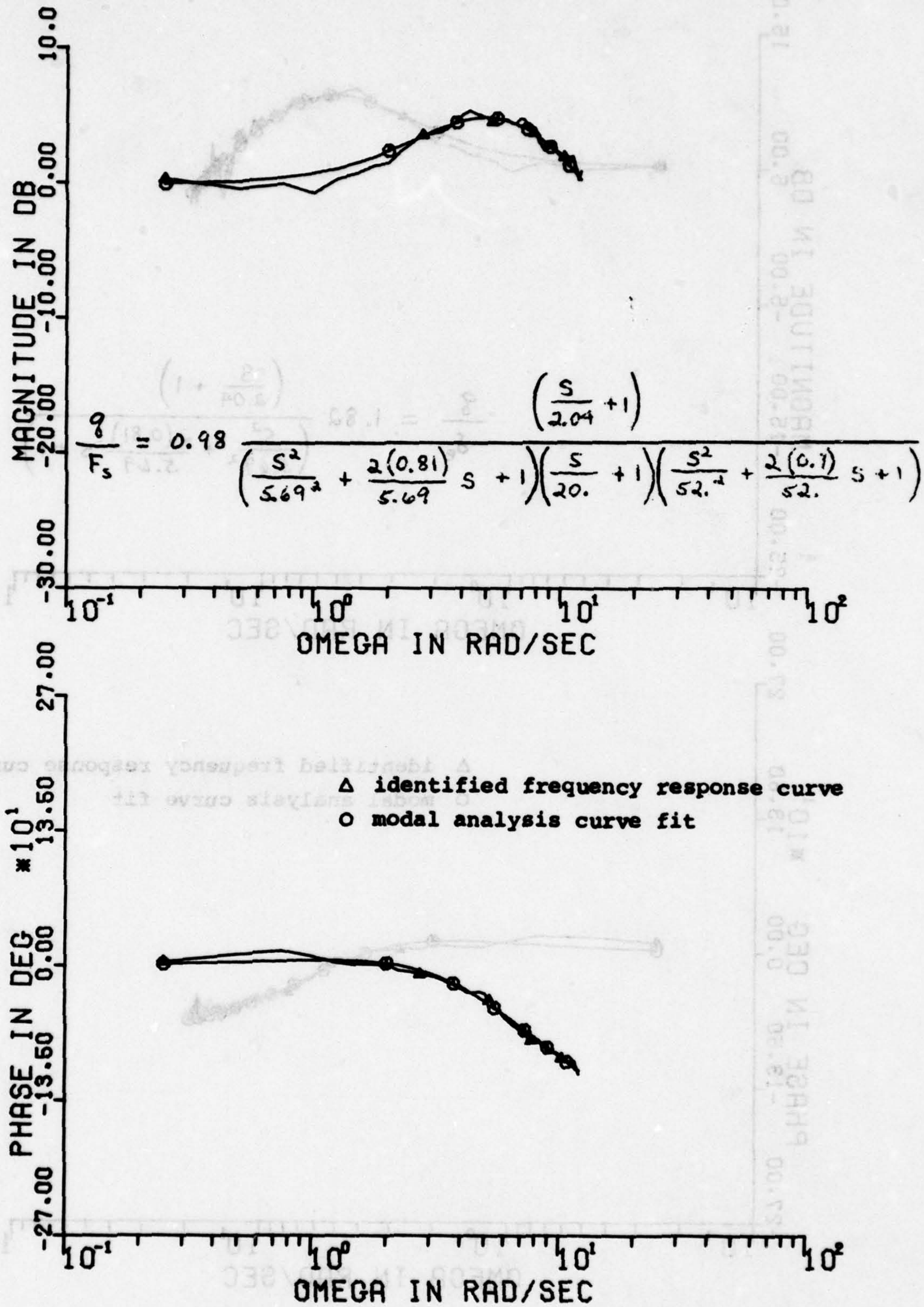


FIGURE 7 RESULTS OF MODAL ANALYSIS OF OVERALL SYSTEM FREQUENCY RESPONSE CURVES

EQUIVALENT SYSTEM MODAL PARAMETER	CASE 1 (FIGURE 8) (0 to 10 radians/sec)		CASE 2 (FIGURE 9) (0 to 10 radians/sec)		CASE 3 (FIGURE 10) (0 to 15 radians/sec)	
	INITIAL USER ESTIMATE	FINAL VALUE	INITIAL USER ESTIMATE	FINAL VALUE	INITIAL USER ESTIMATE	FINAL VALUE
K_E, $\frac{\text{degrees/second}}{\text{pound}}$	0.971	1.217	0.971	0.969	0.971	1.361
a, seconds	-0.05	-0.000077	0.10	0.066	-0.05	-0.000069
$\frac{1}{T_E}$, $\frac{1}{\text{seconds}}$	1.00	9.44	5.00	2.37	1.00	3.32
ω_E, radians/second	6.00	6.52	8.00	5.58	6.00	8.41
ζ_E, non-dimensional	0.70	0.42	0.90	0.71	0.70	0.46
		COST FUNCTION = 0.720		COST FUNCTION = 0.233		COST FUNCTION = 1.728

TABLE 1 RESULTS OF MODAL ANALYSIS OF FREQUENCY RESPONSE CURVES IN FIGURE 4, SHOWING THE INFLUENCE OF THE USER'S INITIAL ESTIMATE AND THE FREQUENCY RANGE OF THE FIT

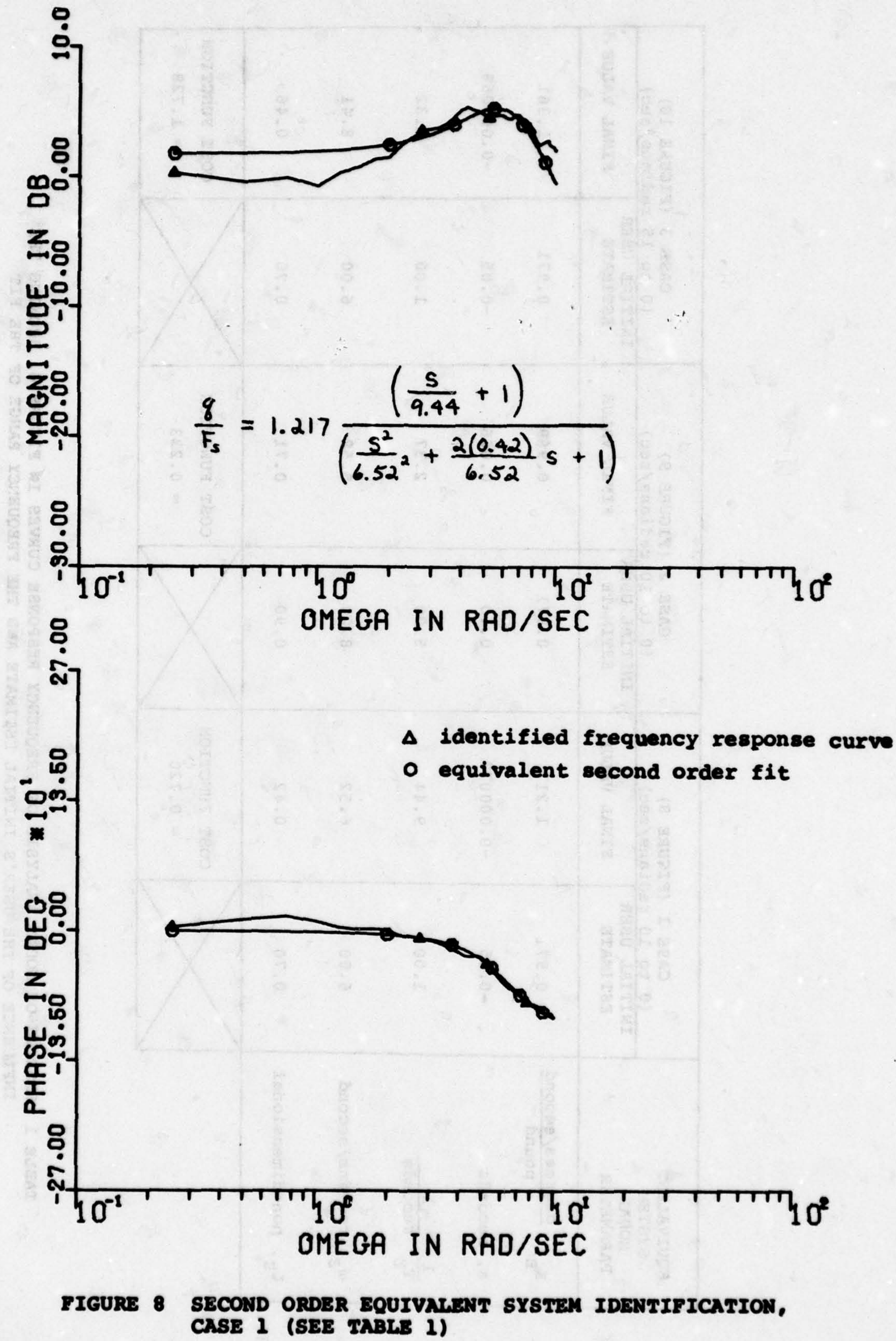


FIGURE 8 SECOND ORDER EQUIVALENT SYSTEM IDENTIFICATION, CASE 1 (SEE TABLE 1)

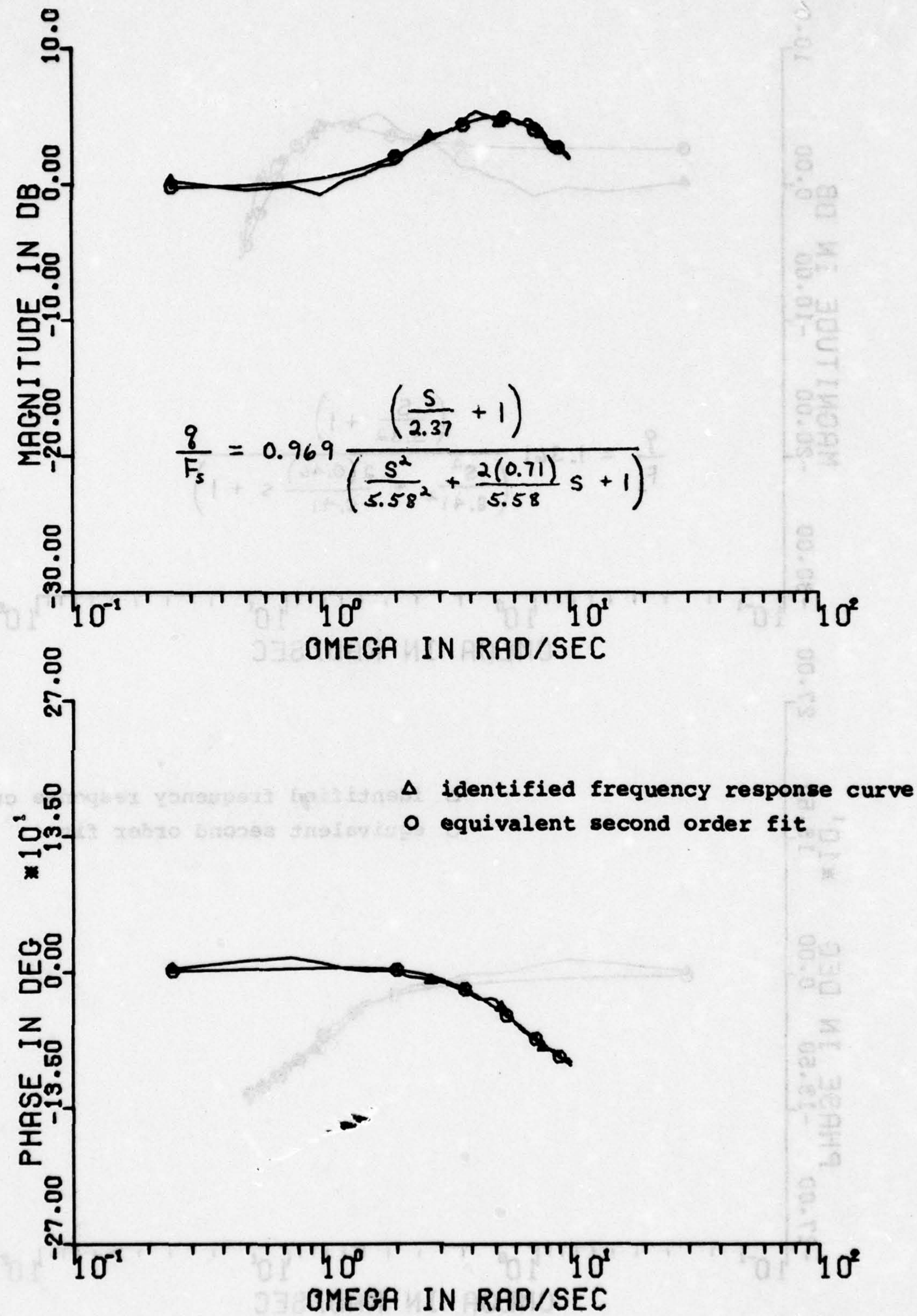


FIGURE 9 SECOND ORDER EQUIVALENT SYSTEM IDENTIFICATION, CASE 2 (SEE TABLE 1)

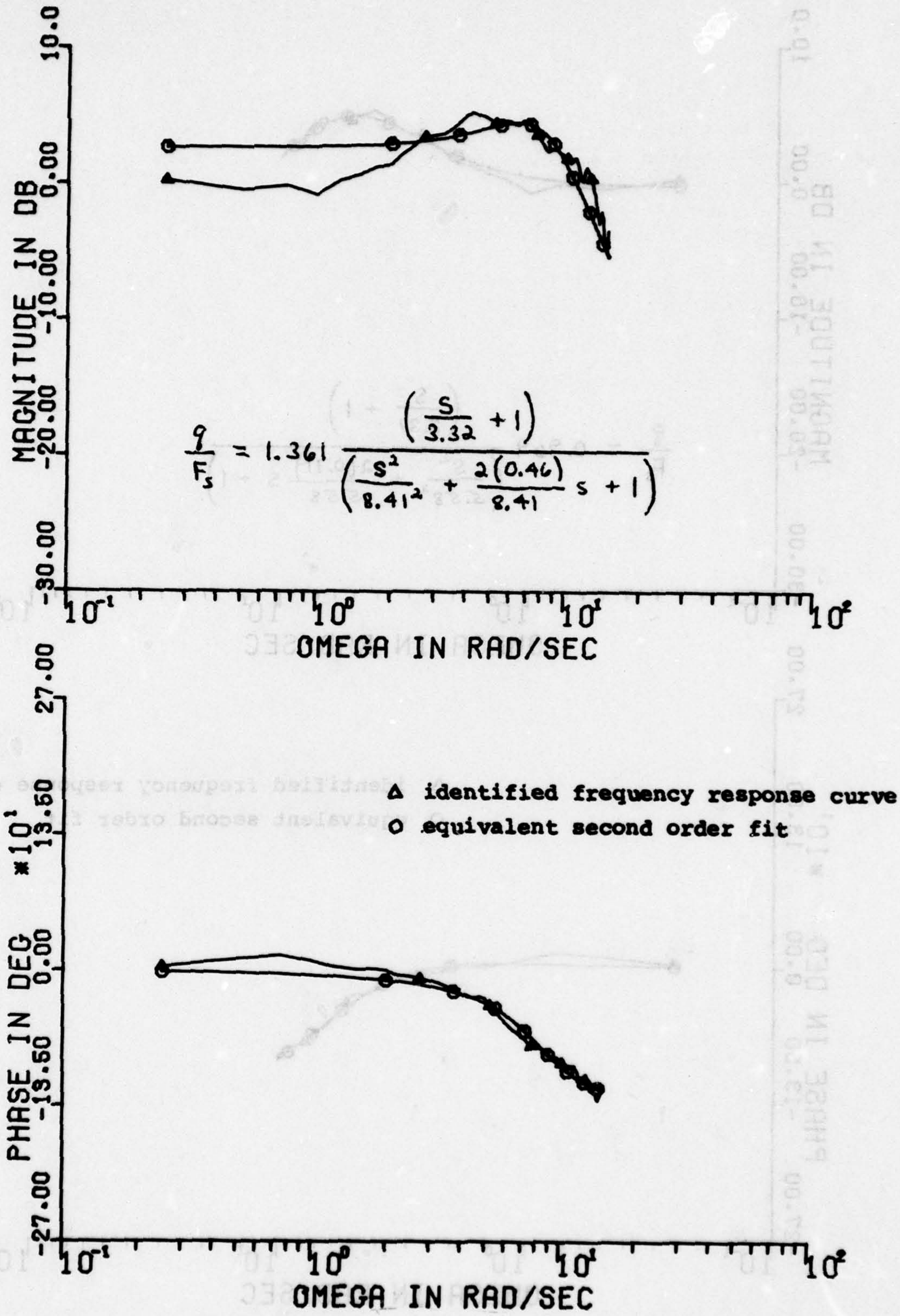


FIGURE 10 SECOND ORDER EQUIVALENT SYSTEM IDENTIFICATION, CASE 3 (SEE TABLE 1)

Ralph Smith, SRL: I'd like to merely comment that it is unfortunate that the equivalent systems work seems to detract from what I believe is the valuable part of your work. Namely, the frequency response measures of airplane response. This is what we need, I believe. I further believe that it's all we need.

EFFECTS OF CONTROL SYSTEM DYNAMICS ON FIGHTER APPROACH AND LANDING LONGITUDINAL FLYING QUALITIES

by

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Air Force Flight Dynamics Laboratory
Dayton, Ohio

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ABSTRACT

The effects of significant control system dynamics on fighter approach and landing longitudinal flying qualities were investigated in flight using the USAF/Calspan variable stability NT-33 aircraft. Two pilots evaluated 49 different combinations of control system and short period dynamics while performing representative approach and landing tasks. The landing task for the majority of the evaluations included an actual touchdown. Pilot rating and comment data, supported by task performance records, indicate that the landing task, in particular the last 50 ft of the task, is clearly the critical task for aircraft with significant control system lags. For these aircraft, a sharp degradation in flying qualities takes place during this critical phase of the landing task; for example, severe pilot induced oscillations occurred during the landing task but were not in evidence during the approach task. The results provide a data base for the development of suitable flying qualities requirements which are applicable to aircraft with significant control system dynamics; the results show that the present landing approach requirements in MIL-F-8785B(ASG) are not adequate; in particular, they are not applicable to aircraft with complex flight control systems.

Section 1
INTRODUCTION AND PURPOSE

In recent years, the demand for increased fighter capability, in combination with the demonstrated reliability of modern electronic systems, has led to more complex flight control systems. For example, the latest fighter aircraft designs include sophisticated digital flight control concepts and revolutionary fly-by-wire flight control systems. The additional complexity of these highly augmented aircraft designs is not a problem in itself; however, significant additional control system dynamics are typically introduced which can potentially alter the flying qualities of the aircraft dramatically. Flying qualities requirements, or control system design criteria, must account for the effects of these additional control system dynamics to be of any value. Unfortunately, response criteria based on classical aircraft characteristics, such as those presented in MIL-F-8785B (Reference 1) are not adequate to evaluate the flying qualities of modern highly augmented fighter aircraft.

A necessary first step in the development of suitable flying qualities evaluation criteria for aircraft with significant control system dynamics was the collection of applicable flying qualities data. In response to this need, a series of research programs was conducted using the USAF/Calspan NT-33A variable stability aircraft (References 2, 3, 4 and 5). These programs concentrated on the longitudinal flying qualities of highly augmented fighter aircraft for maneuvering and tracking tasks (Flight Phase Category A). In particular, pitch maneuver response criteria which consider the total aircraft dynamic system were developed by Neal and Smith (Reference 4) for fighter aircraft performing tracking tasks. The closed-loop pitch attitude tracking criterion of Reference 4 has, in fact, been used with good success as a flying qualities evaluation tool for today's complex fighter aircraft.

In the absence of suitable flying qualities data for the landing approach task (Flight Phase Category C), the concepts developed in Reference 4 were modified somewhat and extrapolated to cover this flight phase (References 6 and 7). Unfortunately, this attempt to provide a suitable

flying qualities criterion for this important flight phase which is applicable to fighter aircraft with significant control system dynamics failed its first real test.

Briefly the story of the test is as follows. Prior to their first flights, both the YF-16 and YF-17 prototypes were simulated in the NT-33A in-flight simulator (Reference 8); these aircraft are both highly augmented and exhibit higher order responses to pilot inputs due to the presence of significant control system dynamics. Of particular interest at this point is the experience in the NT-33A with the landing approach simulation of the YF-17.

- First, the original longitudinal flight control system as simulated in-flight, resulted in very poor flying qualities in the landing approach task, particularly in the final stages of the approach close to the runway.
- Second, this landing problem was not predicted by existing pitch maneuver response criteria, including the extrapolation of the closed-loop Neal/Smith criterion.
- Third, the very poor longitudinal flying qualities in the landing approach were not observed during ground simulation studies on a very sophisticated simulator; however, the deficiencies were dramatically exposed during the initial in-flight simulation sorties.

Control system modifications were then proposed, implemented, and evaluated in the NT-33A until satisfactory longitudinaly flying qualities were observed.

The lessons presented by the YF-17 landing approach simulation experience, in combination with the previous research programs documenting the significant effects of control system dynamics on longitudinal flying qualities, clearly indicated the need for landing approach flying qualities data applicable to highly augmented aircraft. Further, the evidence showed that the tasks must include actual touchdowns. The research program described in this paper was conceived in response to these observations.

The purpose of the research program described in this paper may be summarized as follows:

- To gather pertinent background data on the longitudinal flying qualities of highly augmented fighter aircraft for the landing approach flight phase - including the flare and touchdown (Flight Phase Category C, Class IV.)

- To show whether the flare and touchdown tasks are indeed more demanding than the approach task alone and therefore are the critical landing approach task.

- To lay the groundwork for the development of longitudinal response criteria for the landing approach tasks which are applicable to aircraft with complex control systems, as well as those whose dynamics can be described by classical parameters.

- To gather data on pilot induced oscillations (PIO's) in the landing task with which existing PIO criteria can be evaluated.

The results of the research program using the NT-33A in-flight simulator to study the effects of control system dynamics on landing approach longitudinal flying qualities are summarized in the remainder of this paper. A more detailed account of this research program is presented in the final report, Reference 14.

Section 2
EXPERIMENT DESIGN

2.1 OBJECTIVES

The objective of the in-flight simulation program was to produce an approach and landing longitudinal flying qualities data base from which a suitable response criterion, applicable to highly augmented fighter aircraft, can eventually be developed. Accordingly, the primary evaluation characteristics were selected using the rationale that a broad range of representative aircraft and control system dynamics should be explored rather than specific control system augmentation schemes. In addition, a small portion of the flight program was devoted to the evaluation of configurations with special features; the details of these configurations are discussed in Subsection 2.5. The following subsection is directed at the primary evaluation configurations.

2.2 EXPERIMENT VARIABLES

The block diagram of Figure 2-1 represents how the pilot would view the total longitudinal pitch dynamic "package" which he flies.

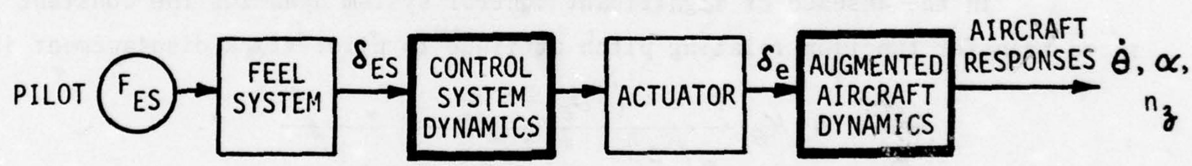


Figure 2-1. LONGITUDINAL RESPONSE BLOCK DIAGRAM

The primary variables in the experiment are the dynamic elements in the heavy blocks: the control system dynamics and the aircraft dynamics. One assumes that we are dealing with aircraft in which the desired augmented aircraft dynamics are achieved, but additional dynamics are introduced-in the form of prefilters, compensation networks, or digital computational

delays - to produce a higher-order system. The term "higher-order system" is used to describe a system with additional significant dynamic modes in addition to the classical short period and phugoid longitudinal response modes. An alternate viewpoint would be that the aircraft dynamics, representing the bare airframe, are combined with prefilter dynamics to produce a higher-order system. The controlled experiment variables may be summarized as follows:

- Aircraft short period dynamics
- Control system dynamics
- Task
  - Full task, including flare and touchdown, or
  - Approach-only task with no touchdown

The details of the aircraft dynamics and control system dynamics selected to form the primary evaluation configurations are discussed in the following subsections.

2.3 AIRCRAFT SHORT PERIOD DYNAMICS

In the absence of significant control system dynamics the constant speed transfer function relating pitch attitude to pilot stick displacement is:

$$\frac{\theta}{\delta_{ES}} = K_\theta \frac{(\tau_{\theta_2} s + 1)}{s\left(\frac{s^2}{\omega_{SP}^2} + \frac{2\zeta_{SP}}{\omega_{SP}} s + 1\right)}$$

Five combinations of ω_{SP} and ζ_{SP} were selected to span fairly wide ranges, relative to the requirements of MIL-F-8785B (Category C). These configurations represent the experiment base configurations; each set of evaluation configurations consists of a base configuration in combination with a variety of control system dynamics. The five base configurations (1-1 through 5-1) are compared with the Category C MIL-F-8785B requirements in Figure 2-2 for the nominal landing flight condition, which is:

- V_{ind} = 120 knots
- V_T = 205 ft/sec
- n_z/α = 4.5 g/rad
- τ_{θ_2} = 1.4 sec
- $1/\tau_{\theta_2}$ ≃ 0.7 rad/sec

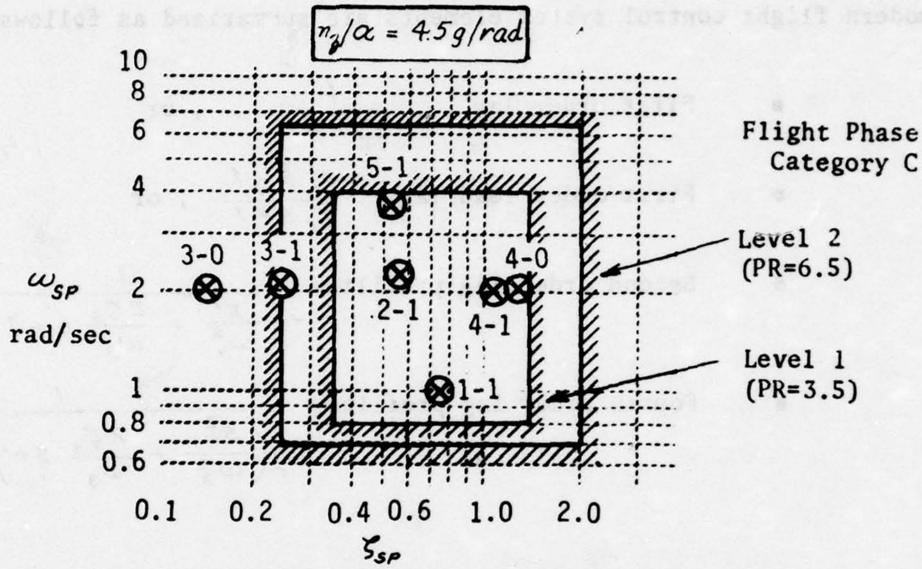


Figure 2-2. COMPARISON OF PRIMARY SHORT PERIOD CONFIGURATIONS WITH MIL-F-8785B

Configurations 3-0 and 4-0, shown in Figure 2-2, represent alternate base configurations which were never evaluated in combination with additional control system dynamics. The very lightly damped configuration (3-0) was specifically selected because it was predicted to be PIO-prone. Along with those configurations in which the addition of control system dynamics produce PIO's, this configuration was intended as an appropriate test of the PIO criterion of Reference 9. Configuration 4-0 was selected as a representative heavily damped configuration but was rejected as a base configuration in the initial phases of the evaluation flying because it was rated unsatisfactory by the pilot.

2.4 CONTROL SYSTEM DYNAMICS

Each base short period configuration was evaluated in combination with a variety of representative control system dynamics. The various forms of the control system dynamics selected which are representative of typical modern flight control system elements are summarized as follows:

- First order lag, $\dfrac{1}{\tau_2 s + 1}$, or
- First order lead/lag, $\dfrac{\tau_1 s + 1}{\tau_2 s + 1}$, or
- Second order lag prefilter $\dfrac{1}{\dfrac{s^2}{\omega_3^2} + \dfrac{2\zeta_3}{\omega_3} s + 1}$, or
- Fourth order lag prefilter $\dfrac{1}{\left(\dfrac{s^2}{\omega_3^2} + \dfrac{2\zeta_3}{\omega_3} s + 1\right)\left(\dfrac{s^2}{\omega_4^2} + \dfrac{2\zeta_4}{\omega_4} s + 1\right)}$

2.5 ADDITIONAL EVALUATION CONFIGURATIONS

Since the approach and landing results of the YF-17 simulation program conducted in the NT-33 (Reference 8) represent such a significant example of the effects of control system dynamics on longitudinal flying qualities, the previously simulated original and modified YF-17 configurations were selected for evaluation in this program. The original YF-17 landing case also represents an excellent data point for testing the PIO criterion suggested in Reference 9. In addition, inclusion of this case affords the opportunity to evaluate thoroughly a significant anomaly in the world of simulation: the extreme PIO problem was not observed during ground simulation studies but was clearly evident in the in-flight simulation. The configurations are identified as:

- 6-1: YF-17 original control system
- 6-2: YF-17 modified control system

In support of suggested revisions to MIL-F-8785B in the area of longitudinal static instability, three statically unstable configurations were evaluated. Essentially these configurations are Configuration 2-1 with the center of gravity moved aft of the neutral point ($M_\alpha > 0$). These configurations are representative of failure states which could possibly occur in highly augmented aircraft like the F-16 which operate at negative static margins. If the angle of attack feedback should fail in this condition the pilot could be faced with landing a statically unstable aircraft. These configurations are identified as:

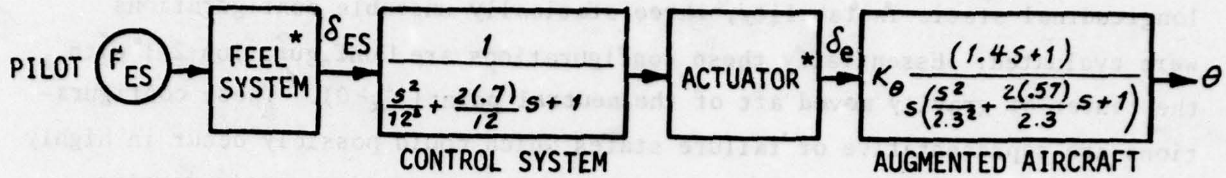
- 7-1, 7-2, 7-3

2.6 CONFIGURATION SUMMARY

A total of 49 configurations were evaluated: 44 basic control system/short period configurations and 5 additional configurations, as summarized in Figures 2-3 and 2-4.

The existing Neal/Smith longitudinal flying qualities evaluation criterion (Reference 4) and the results of a study of the YF-16, YF-17 simulation program (Reference 10) were used to guide the initial configuration selection; the digital computer version of the criterion developed by Mayhew (Reference 11) was used in this process.

Before attempting to interpret the configuration summaries in Figures 2-3 and 2-4, a brief review is in order. A typical evaluation configuration consists of the dynamic elements of the feel system, control system, aircraft and actuator in series. For example, the constant speed transfer function of pitch attitude response to pilot stick force for Configuration 2-7 is:



*Feel system and actuator dynamics were fixed; details are in Subsection 2.7.

The value of K_θ is a function of the elevator gearing selected by the evaluation pilot as discussed in Subsection 2.8.

The CONTROL SYSTEM and AIRCRAFT DYNAMICS for the primary evaluation configurations are summarized in Figure 2-3; the important characteristics of the additional configurations are summarized in Figure 2-4. Remember that the total configuration dynamic model for each configuration includes the fixed dynamic contribution of the feel system and the actuator.

For the majority of the configurations (1 through 6), the phugiod, or long term, response characteristics are those of the NT-33 as modified somewhat by the longitudinal feedback gains used to achieve the short period dynamics. Details are given in Reference 14. From the flight path control viewpoint, all the evaluations were on the "front side" of the power required versus drag curve.

CONTROL SYSTEM DYNAMICS				SHORT PERIOD DYNAMICS (Nominal) V_{ind} = 120 Kt; n_z/α = 4.5 g/rad; τ_{θ_2} = 1.4 sec; ω_{sp}/ζ_{sp}				
τ_1	τ_2	ω_3/ζ_3	ω_4/ζ_4	1.0/.74	2.3/.57	2.2/.25	2.0/1.06	3.9/.54
0.4	0.1	-	-	1-A	2-A			
0.3	0.1	-	-	1-B				
0.2	0.1	-	-	1-C	2-C	3-C	4-C	
0	0	-	-	1-1	2-1	3-1(3-0)*	4-1(4-0)*	5-1
	0.1	-	-	1-2	2-2	3-2		
	0.25	-	-	1-3	2-3	3-3	4-3	5-3
	0.5	-	-	1-4	2-4		4-4	5-4
	1.0	-	-					5-5
	0	16/.7	-	1-6	2-6	3-6	4-6	5-6
		12/.7	-		2-7	3-7	4-7	5-7
		9/.7	-	1-8				
		6/.7	-		2-9			
		4/.7	-		2-10		4-10	
0	0	16/.93	16/.38	1-11	2-11		4-11	5-11

*ω_{sp}/ζ_{sp} for Configuration 3-0 is 2.1/.14; for Configuration 4-0, 2.1/1.23

NOTES:
- First number indicates base aircraft configuration simulated; second number or letter identifies control system dynamics; letters for control system lead; numbers for lag.
- Total configuration dynamic model includes feel system and actuator dynamics (see Subsection 2.7).

Figure 2-3. SUMMARY OF PRIMARY EVALUATION CONFIGURATIONS

CONFIGURATION	CONTROL SYSTEM DYNAMICS	ω_{sp} / ζ_{sp}
6-1 (YF-17 Original)	$\dfrac{(.5s+1)(.43s+1)}{(.2s+1)(1.1s+1)\left(\dfrac{s^2}{4^2}+\dfrac{2(.7)}{4}s+1\right)}$	1.9 /.65
6-2 (YF-17 Modified)	$\dfrac{(.5s+1)(.43s+1)(.06s+1)}{(.2s+1)(.1s+1)(1.1s+1)}$	1.9/.65
	Time to Double Amplitude, T_d (Sec)	
7-1	$\simeq 6$	
7-2	$\simeq 4$	
7-3	$\simeq 2$	

NOTES: • Total configuration dynamic model includes feel system and actuator dynamics (see Subsection 2.7).

Figure 2-4. SUMMARY OF ADDITIONAL EVALUATION CONFIGURATIONS

2.7 PITCH FEEL SYSTEM AND ACTUATOR CHARACTERISTICS

The feel system characteristics were held fixed for all the configurations evaluated in the program; a representative spring gradient of 8 lb/in was chosen and essentially zero breakout or friction forces were present. The feel system transfer function is:

$$\frac{\delta_{ES}}{F_{ES}} = \frac{.125}{\left(\dfrac{s^2}{26^2} + \dfrac{2(.6)s}{26} + 1\right)} \quad (in./lb)$$

The NT-33 pitch actuator characteristics were essentially constant for all configurations with the following values:

$$\omega_a = 75 \text{ rad/sec}$$

$$\zeta_a = 0.7$$

2.8 PITCH CONTROL SENSITIVITY

The gearing ratio between the elevator and the stick position was selected by the pilot for each flight evaluation of a configuration, as discussed in more detail in Section 3. Ideally each dynamic configuration should have been evaluated with several values of gearing ratio, but this procedure was beyond the scope of this flight program.

Section 3
CONDUCT OF THE EXPERIMENT

The control system and aircraft dynamics discussed in Section 2 were mechanized in the USAF variable stability NT-33, operated by Calspan (see Figure 3-1) and described in Reference 12.

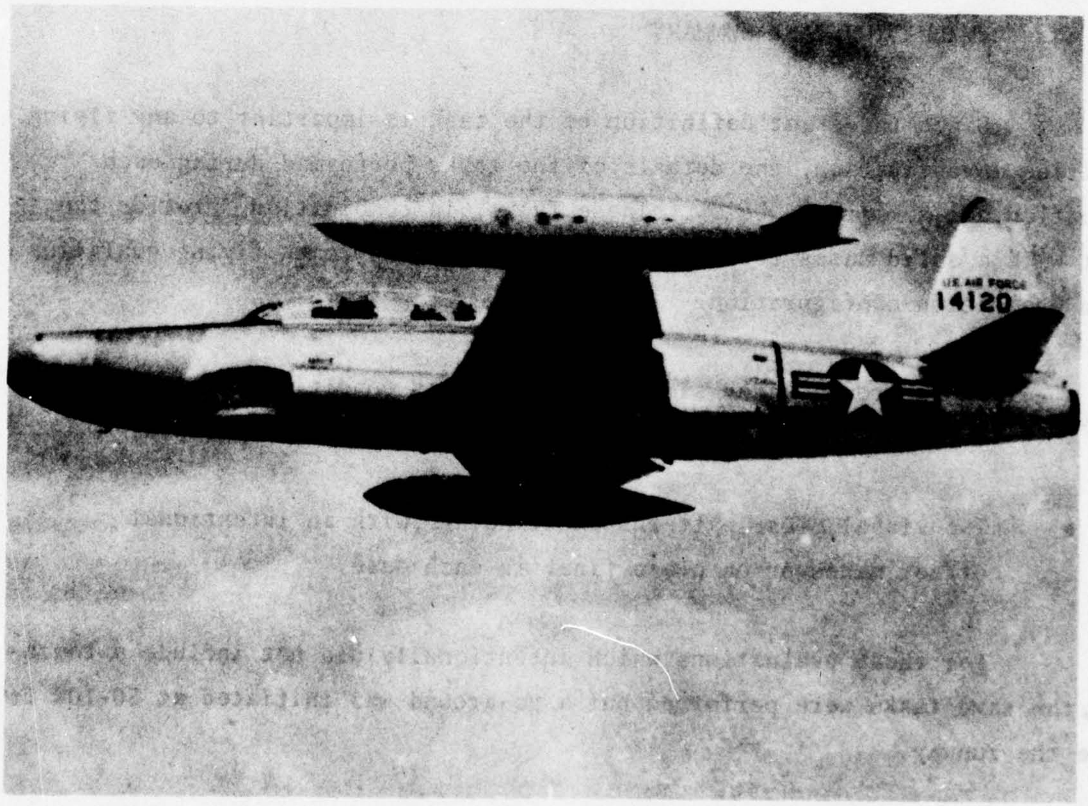


Figure 3-1. USAF/CALSPAN VARIABLE STABILITY NT-33 AIRCRAFT

3.1 SIMULATION SITUATION

For this program, the simulated aircraft was defined as an all-weather, single seat, fighter aircraft (Class IV). The pilot was therefore required to extrapolate to this environment which would include additional duties such as navigation and communication.

Since inclusion of wind and turbulence as controlled variables was beyond the limited scale of this program, flights were, of necessity, conducted in a wide range of wind and turbulence; conditions encountered are considered normal for typical fighter operations. The pilots were asked to evaluate the aircraft in the conditions of the day, but to comment, if desired, on the projected effects of different wind and turbulence conditions.

3.2 EVALUATION TASK SUMMARY

Since the exact definition of the task is important to any flying qualities investigation, the details of the tasks performed during each evaluation are summarized below. These tasks, in combination, provide the pilot with a solid basis for assessing the landing approach flying qualities of an evaluation configuration.

- ILS approach under simulated instrument conditions, down to 200 ft above runway, followed by a visual landing, plus

- Two visual close patterns and landings, with an intentional offset maneuver on close final in each case.

For those evaluations which intentionally did not include a touch-down, the same tasks were performed but a go-around was initiated at 50-100 ft above the runway.

Great care was taken to ensure that the evaluation pilots performed these tasks in a realistic fashion. For example, they were instructed to consider each approach and landing as a final "must land" situation; the 500 ft touchdown zone on the 9100 ft runway was clearly marked on the runway, beginning approximately 1500 ft from the threshold for flight safety reasons. The pilots were not allowed by the safety pilot to back out of the task and let the touchdown point drift down the runway in an unrealistic fashion. These instructions were not interpreted by the pilots to mean that the task was treated as an unrealistic "game" demanding unrealistic precision on the part of the pilots. Touchdown with normal sink rates could be made in the NT-33.

3.3 EXPERIMENT DATA

The data from the experiment take three forms: pilot ratings, pilot comments, and records of task performance, including the discrete error tracking tasks. Examples of the performance records are presented in Appendix II. The pilot ratings and comments are clearly tied together and should not be viewed as separate data. At the completion of the evaluation tasks, the pilot was asked to assign an overall pilot rating using the Cooper-Harper Rating Scale (Reference 13). In addition, for the evaluations which included the complete landing task, the pilot was asked to give a separate rating for the approach task alone (down to approximately 50 feet above the runway). Only during the last half of the program did the pilots feel confident enough to give both ratings. The pilots were asked to assign the ratings before making detailed comments since their task performance was then fresh in their minds.

After the initial ratings, the pilot was asked to make recorded comments on specific items listed in the Pilot Comment Card.

3.4 EVALUATION SUMMARY

The two pilots performed a total of 83 flight evaluations of the 49 different configurations during the program requiring 24 flights of approximately 1½ hours each. The distribution of evaluations and flights between the pilots is as follows:

	PILOT A	PILOT B
Flights:	17	7
Evaluations with Landings:	51	21
Evaluation with Low-Approach Only:	8	3

Section 4
EXPERIMENT RESULTS

The results of the experiment described in the preceding sections are in the form of pilot ratings, comments and records of task performance. Since a complete analysis of the data and the development of appropriate design criteria or flying qualities requirements is clearly beyond the scope of this program, the discussion of the results in this section is centered on the pilot rating and comment data; a limited discussion of the applicability of the Neal/Smith closed-loop pitch attitude tracking criterion is, however, included.

A summary of the pilot ratings for each configuration is presented in Table 4-1.

4.1 CORRELATION WITH MIL-8785B

The overall pilot ratings for the base configurations from each set of configurations - those with no significant additional control system dynamics - are compared in Figure 4-1 with the ω_{SP}, ζ_{SP} Category C boundaries from MIL-F-8785B (Reference 1). For these comparisons, the nominal 120 KIAS data are used.

TABLE 4-1: PILOT RATING DATA SUMMARY

Config. No.	Aircraft(1) ω_{sp} / ζ_{sp}	Control System(2) τ_1	τ_2	ω_3	PILOT RATINGS OVERALL(4) PILOT A	PILOT B	APPROACH(3) PILOT A	PILOT B
1-A	1.0/.74	0.4	0.1	-	6(4TD)			
B		0.3	0.1	-	5			
C		0.2	0.1	-	4	4		
1-1		-	-	-	4	4		2
-2		-	0.1	-	5			
-3		-	0.25	-	9	10	6*	6*
-4		-	0.5	-	10			
-6		-	-	16	5		5*,2	
-8		-	-	9		8		5
-11		-	-	16(4th)		9		5
2-A	2.3/.57	0.4	0.1	-	4	6	2	3
-C		0.2	0.1	-	1½,4,3,1½		3	
2-1		-	-	-	2	2	3*	
-2		-	0.1	-	4½	4		
-3		-	0.25	-	6			
-4		-	0.5	-	9		5*	3*
-6		-	-	16	5		1½	
-7		-	-	12	7	6	4	3
-9		-	-	6		10		5
-10		-	-	4	10			
-11		-	-	16(4th)		8		4
3-C	2.2/.25	0.2	0.1	-	2	5(4TD)		5
3-0	2.1/.14	-	-	-	5,4			
3-1	2.2/.25	-	-	-	5(4TD)4	7(5TD)	5	7
-2		-	0.1	-	7		6*	
-3		-	0.25	-	10		7*	
-6		-	-	16	7	6	5	5
-7		-	-	12	8		5	

TABLE 4-1: PILOT RATING DATA SUMMARY (CONT.)

Config. No.	Aircraft(1) ω_{sp}/ζ_{sp}	Control System(2) τ_1	τ_2	ω_s	PILOT RATING OVERALL(4) PILOT A	PILOT B	APPROACH(3) PILOT A	PILOT B
4-C	2.0 /1.06	0.2	0.1	-	3	3	1½	2
4-0	2.1/1.23	-	-	-	6			
4-1	2.0/1.06	-	-	-	2			
-3		-	0.25	-	5,7	8	2	5
-4		-	0.5	-	7	6	5*,4	3
-6		-	-	16	4		1½	
-7		-	-	12	3		3	
-10		-	-	4	9		6	
-11		-	-	16(4th)	8		3	
5-1	3.9/.54	-	-	-	7	5		4
-3		-	0.25	-	4½,8	6	4*,3	3*,2
-4		-	0.5	-	6		2	
-5		-	1.0	-	7		2	
-6		-	-	16		6		3
-7		-	-	12	6		2	
-11		-	-	16(4th)	7		3	
6-1	1.9/.65	Unmodified YF-17			10			
-2		Modified YF-17			2			
	T_d (Sec)							
7-1	≃ 6	-	-	-	4			
-2	≃ 4	-	-	-	3			
-3	≃ 2	-	-	-	4	6(3TD)	2	6

NOTES: (1) Aircraft dynamics are for 120 KIAS nominal case:

V_T = 205 ft/sec; n_z/α = 4.5 g/rad; τ_{θ_2} = 1.4 secs.

(2) Complete control system includes feel system and actuator dynamics (see Subsection 2.7); 16 rad/sec fourth order prefilter designated: 16(4th)

(3) Asterisk (*) indicates evaluations for low-approach only task

(4) TD stands for "touchdown"; pilot rating better for landing than for approach in these cases.

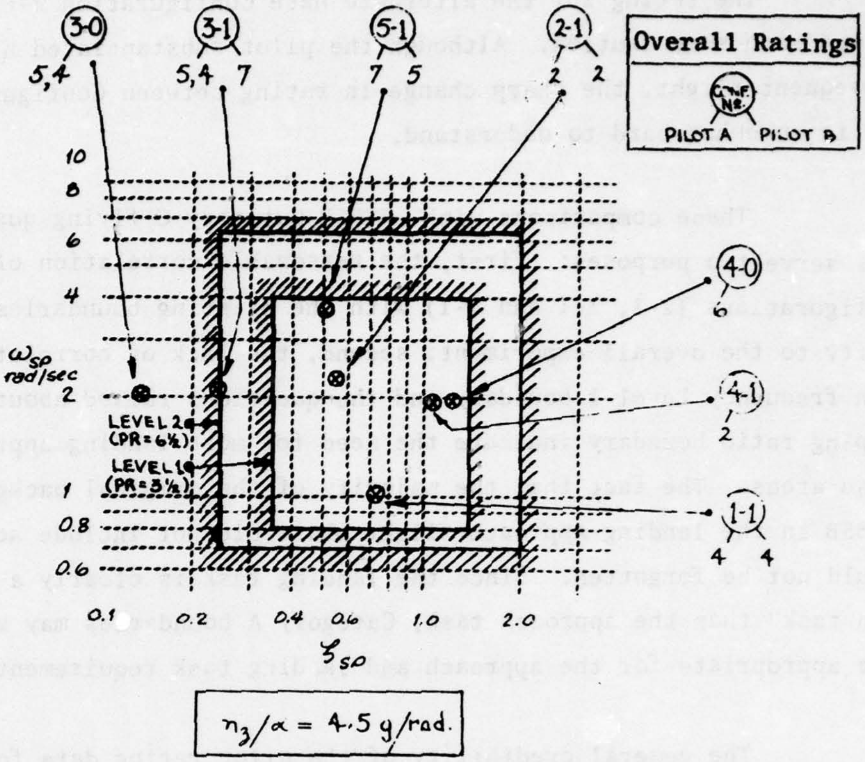


Figure 4-1. CORRELATION OF BASE CONFIGURATIONS WITH MIL-F-8785 (CATEGORY C)

Although the quantity of data is hardly enough to allow definitive comments, the following comments can be made. The pilot ratings for Configurations 3-1 and 4-1 agree reasonably well with the -8785B Level 1 boundaries, while the ratings for Configuration 1-1 indicate that the lower frequency Level 1 boundary is too lenient. For Configuration 3-1, and the alternate base Configuration 3-0, the ratings are somewhat less severe than the -8785B boundaries would predict. Since all of these ratings were obtained in relatively smooth air and turbulence effects would certainly degrade these configurations, the correlation is considered reasonable. The results for Configuration 5-1 indicate that the Level 1 upper boundary on ω_{sp} is too lenient. Although the configuration is likely only of academic interest since aircraft are typically low frequency in the landing approach, the boundary in -8785B does appear to be suspect. Evaluation of this configuration in moderate turbulence would further emphasize the lack of correlation.

The rating for the alternate base Configuration 4-0 should be viewed with some caution. Although the pilot substantiated his rating on a subsequent flight, the sharp change in rating between Configurations 4-1 and 4-0 is somewhat hard to understand.

These comparisons with -8785B Category C flying qualities boundaries serve two purposes: first, the reasonable correlation of several of the configurations (2-1, 3-1 and 4-1) with the existing boundaries lends credibility to the overall experiment; second, the lack of correlation with the high frequency Level 1 boundary and the questions raised about the high damping ratio boundary indicate the need for more landing approach data in these areas. The fact that the majority of the original background data for -8785B in the landing approach flight phase did not include actual landings should not be forgotten. Since the landing task is clearly a "higher pilot gain task" than the approach task, Category A boundaries may very well be more appropriate for the approach and landing task requirements.

The general credibility of the pilot rating data for the base configurations provides a solid base from which to view the remainder of the pilot rating data for these same configurations evaluated with significant additional control system dynamics.

MIL-F-8785B presently contains a requirement which is intended to place limits on control system dynamics by restricting the phase lag, at the short period frequency, between the stick force input and the control surface response. Although the Category A substantiation data (from Reference 3) used for the requirement were not really applicable to the landing approach task (Flight Phase Category C), the requirement applies to this flight phase. In light of the observations from this experiment, this previously unfounded extrapolation has some merit since the two tasks are not apparently that different. The original data suggested a limit of 30 deg of phase lag for Level 1 flying qualities; as shown in Reference 4, this requirement, in its present form, does not apply to aircraft with significant control system

dynamic elements whose characteristics frequencies are close to the aircraft's short period frequency. The results from this experiment corroborate this finding; for example, consider Configurations 1-2 and 2-11. In each case the phase lag of the control system at the short period frequency is on the order of 30 deg, yet the pilot ratings are Level 3.

As observed in the previous control system dynamics experiment in the NT-33 (Reference 4), the pilot evaluates the total response of the aircraft to his inputs and is not concerned with, or even aware of, the characteristics of the individual dynamic elements which combine to produce that response. The step response time histories for the evaluation configuations are presented in Appendix III and illustrate the effects of the various types of control system dynamics evaluated. The high frequency elements, such as the "-6" cases, effectively preserve the shape of the short period response but introduce a transport time delay; while the low frequency elements, such as the "-4" cases, significantly alter the shape of the response to pilot inputs.

Requirements are obviously needed for the landing flight phase which are based on the characteristics of the total response and are not dependent on identifying the response with certain modes of motion, such as the short period response.

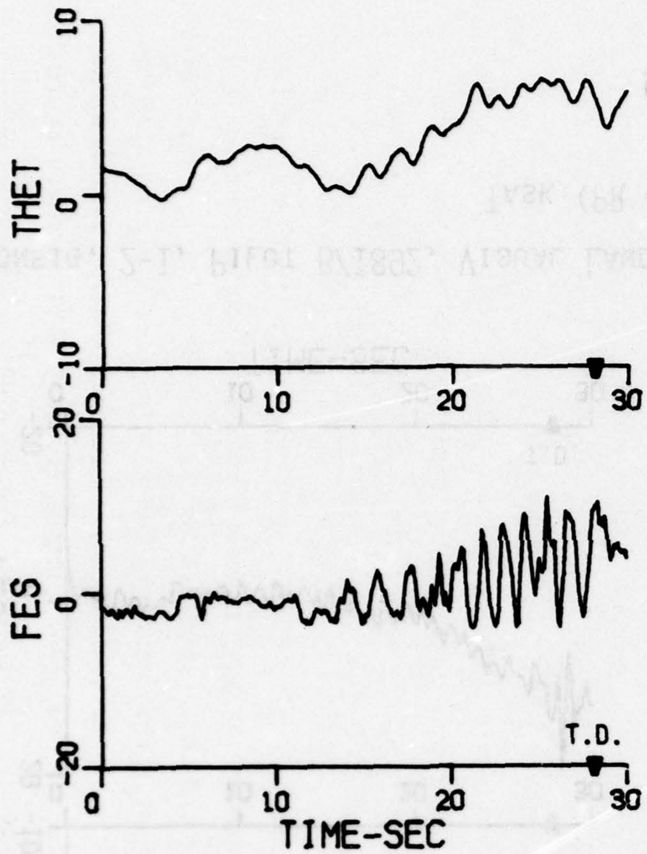
4.2 THE CRITICAL TASK: FLARE AND TOUCHDOWN

One of the objectives of the experiment was to determine whether the final stage of the approach and landing task - the flare and touchdown - is the critical piloting task. Evidence from the simulation of the YF-17 prototype with the original control system (Reference 12) suggested that the major pitch flying qualities problems occurred close to touchdown.

The pilot rating data for the majority of the configurations clearly indicate that the landing task, which means the last 50 ft to touchdown, is the critical task. For the approach task, that is down to 50 ft above the runway, the pilot ratings are typically better than for the overall task which includes the landing task; the difference is dramatic when significant additional control system dynamics are present. Aircraft with good longitudinal flying qualities, such as Configurations 2-C, 2-1, and 4-C, show little difference in the pilot ratings for approach alone versus the overall task with a landing. On the other hand, aircraft with significant additional lag dynamics in the control system, such as Configurations 2-4 and 4-10, show significant differences between the approach alone and overall pilot ratings. The more stringent flare and touchdown task exposes the "flying qualities cliffs" hidden in these aircraft.

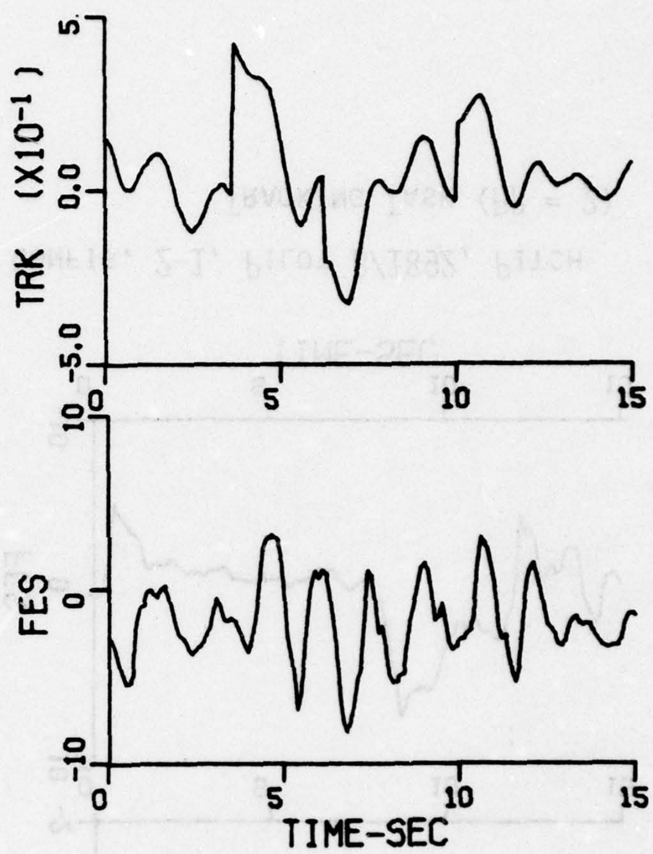
For those approach pilot ratings marked with an asterisk, a go-around was initiated 50 to 100 ft above the runway. The other approach ratings represent the pilot's assessment of the approach portion of the overall task while performing the complete touchdown task. The approach-only ratings confirm the same differences between the approach task and the landing task observed previously, except that the approach pilot ratings for a configuration tend to be worse when only the approach was performed. In general, this difference is not significant except for the approach-only rating given to Configuration 1-6 (PR=5). The configuration received a PR=2 when evaluated in the total task later on the same flight; this variation in rating on the same flight tends to reduce the significance of this rating anomaly.

The critical nature of the last 50 ft of the task environment is dramatically illustrated in the task performance time histories for Configurations 2-4 and 6-1 in Figures 4-2 and 4-5, where a PIO suddenly develops at this point in the task. For contrast, the task performance time histories for Configurations 2-1 and 6-2, without the additional control system dynamics are presented in Figures 4-3 and 4-4. On the figures, THET is pitch attitude in degrees and FES is longitudinal stick force in pounds; TRK is the pitch



CONFIG. 2-4, $\left(\frac{1}{.5\ S+1}\right)$, PILOT A/1888,

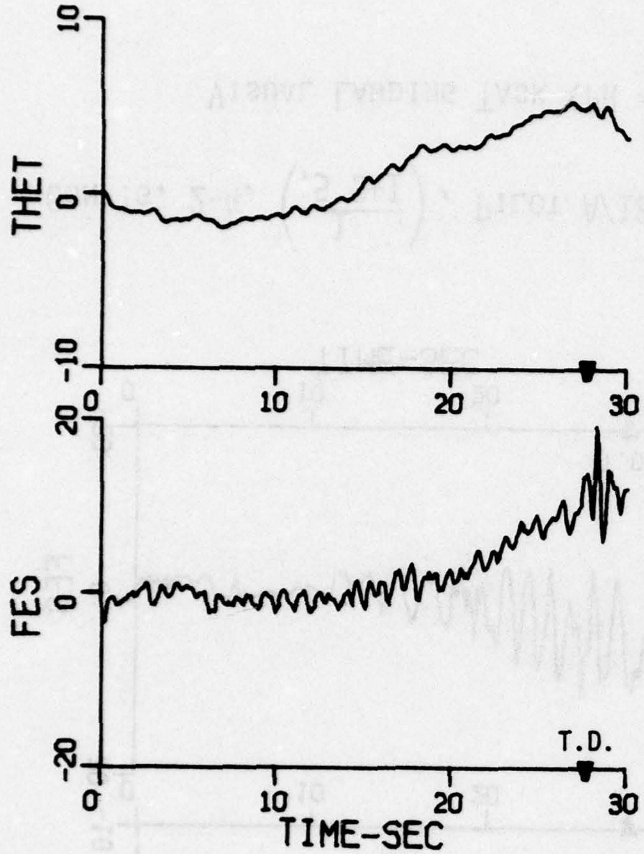
VISUAL LANDING TASK (PR = 9)



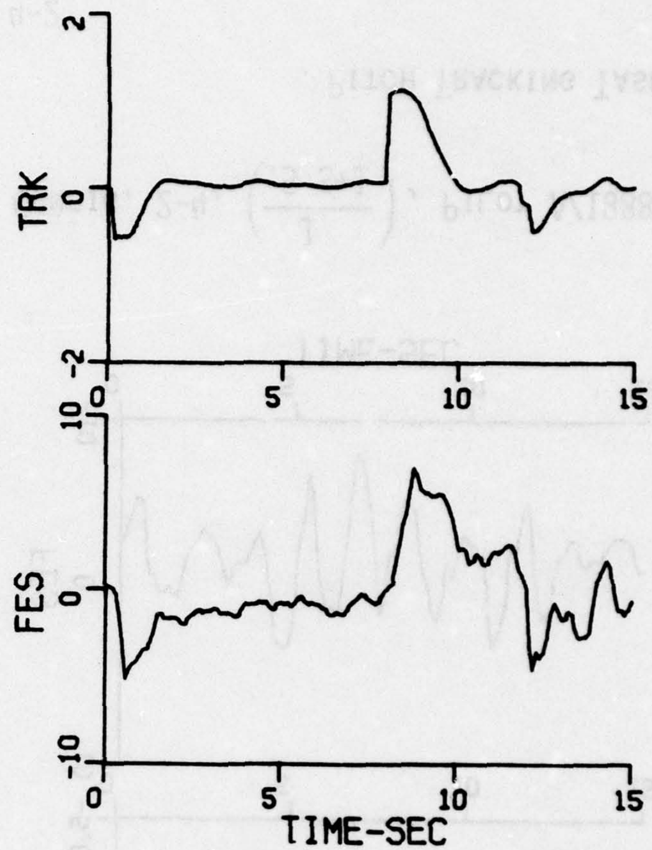
CONFIG. 2-4, $\left(\frac{1}{.5\ S+1}\right)$, PILOT A/1888,

PITCH TRACKING TASK

FIGURE 4-2

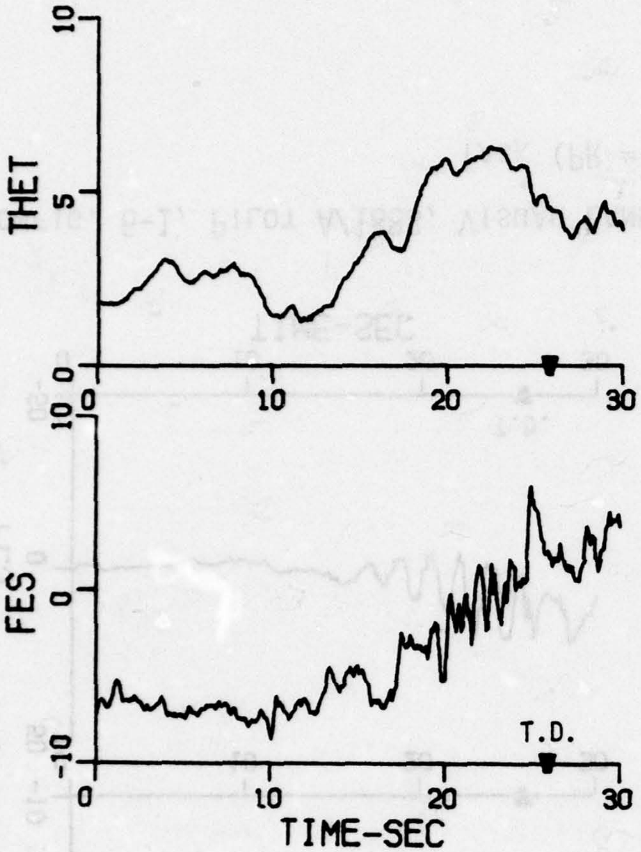


Config. 2-1, Pilot B/1892, Visual Landing Task (PR = 2)

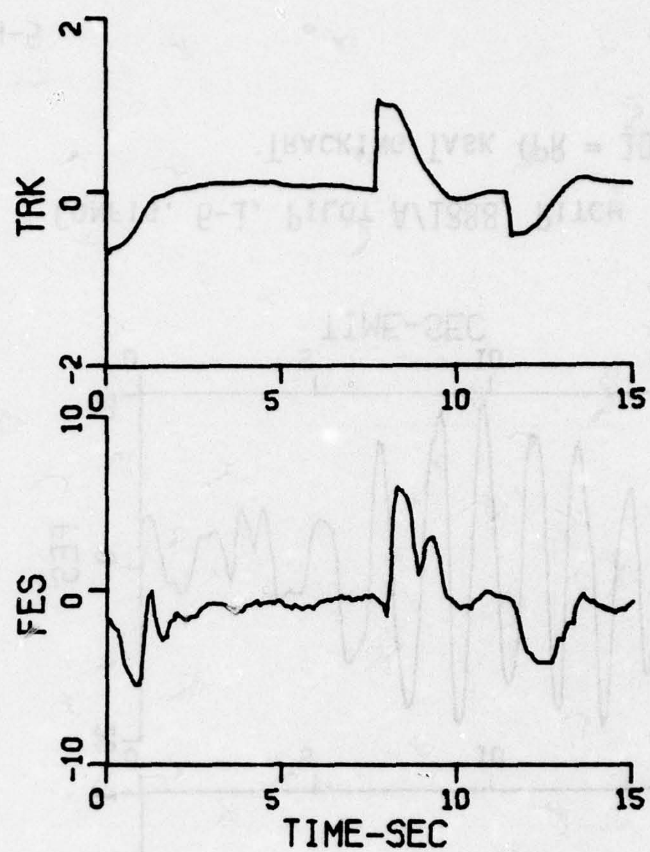


Config. 2-1, Pilot B/1892, Pitch Tracking Task (PR = 2)

Figure 4-3

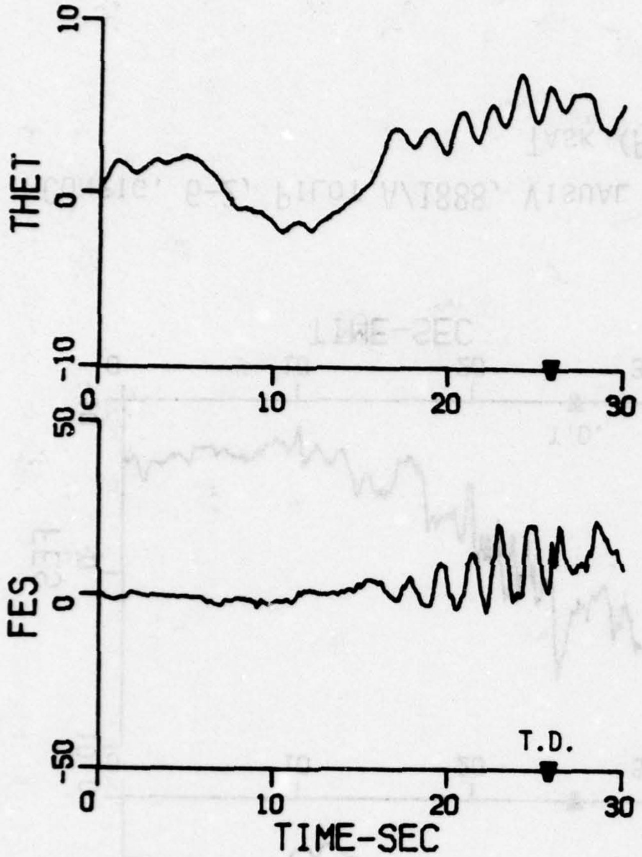


Config. 6-2, Pilot A/1888, Visual Landing Task (PR = 2)

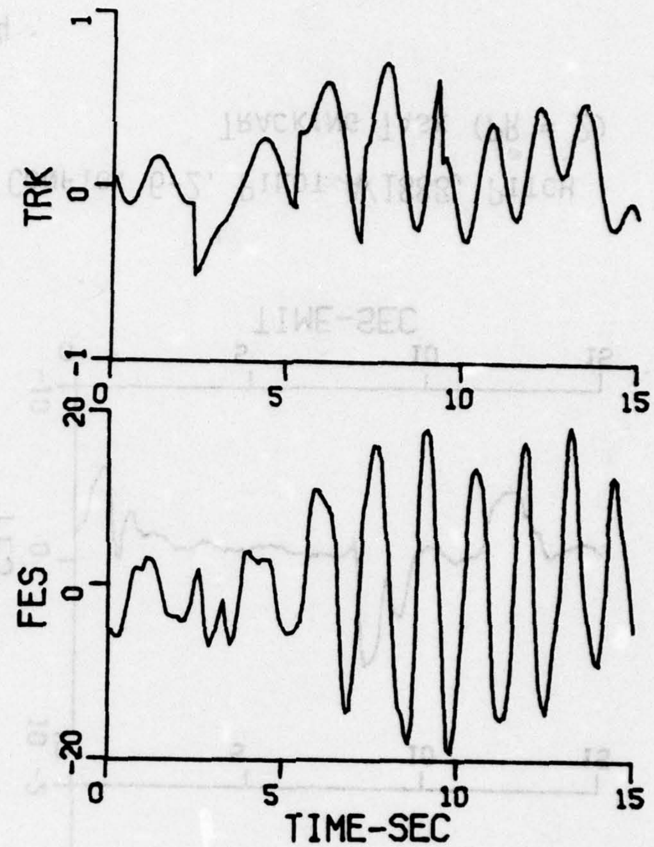


Config. 6-2, Pilot A/1888, Pitch Tracking Task (PR = 2)

Figure 4-4



CONFIG. 6-1, PILOT A/1888, VISUAL LANDING TASK (PR = 10)



CONFIG. 6-1, PILOT A/1888, PITCH TRACKING TASK (PR = 10)

FIGURE 4-5

attitude error on the tracking needle of a discrete error tracking task which the pilot performed after each evaluation, (see Reference 14 for details). Note the scale changes in stick force when comparing configurations.

Configurations 1-A, 3-0, 3-1, 3-C, 7-1, 7-2 and 7-3 represent exceptions to the observation that the landing task is the more critical task. For these configurations, which are essentially without significant control system dynamics, the pilots often commented that the flare and touchdown task was easier than the approach task. The pilot could fly the aircraft better in the flare than on the approach for reasons which further analysis will hopefully expose. All of these configurations, except 1-A, have a common feature: the basic aircraft has a major problem. Configuration 3 has a lightly damped short period response, while Configuration 7-1 through 3 are statically unstable. It could be that the initial response for these configurations is quick enough to allow the pilot to perform the exacting, fighter closed-loop, landing task more accurately than he can the approach task. During the less demanding approach task, the basic aircraft problems are more apparent and therefore annoy the pilot.

The evidence from this experiment indicates that the landing task is clearly different and generally more difficult than the approach task. In the landing task environment the pilot flies differently than on the approach - the experiment results, in general, show that he has a different standard of performance. The flying qualities of a configuration with a poor combination of dynamics can be degraded significantly by the demands of the landing task.

4.3 EQUIVALENT TRANSPORT TIME DELAYS

Since the use of digital flight control systems is now a reality, it is important to understand the impact on flying qualities of the transport time delays associated with the necessary digital computations. Unfortunately, exact time delays, i.e. not lags but delays during which no response occurs, were not included in this experiment. However, an equivalent time delay can be estimated for the high frequency control system elements evaluated in the

program. These elements, such as "-6" and "-11" for example, introduce phase lag but do not affect the amplitude of the response for frequencies near the relatively lower frequency short period. The effect of these higher frequency prefilters is therefore similar to a pure time delay ($e^{-\tau s}$) for frequencies much lower than the prefilter natural frequency.

For reference, the equivalent time delays, in millisecs (ms), for the prefilters simulated are:

ω_n	Control System Element	τ_{equiv} (ms)
16	"-6"	90
12	"-7"	120
9	"-8"	160
6	"-9"	230
4	"-10"	250
16(4th)	"-11"	165
26	Feel System	45
75	Elevator Actuator	20

4.4 CONFIGURATION 7: STATICALLY UNSTABLE CASES

These configurations were included in the evaluation matrix as a mini-experiment to gain some insight into the effects of static instabilities on landing flying qualities. The increased capabilities of modern fly-by-wire flight control system designs enable fighter aircraft to operate at more efficient aft c.g. conditions. However, this condition means that the unaugmented aircraft is statically unstable. An obvious question which then arises is: If part of the augmentation system should fail and the pilot is left with a statically unstable vehicle, can he land it safely?

The evaluation results for these configurations are rather startling in that the pilots could perform the landing task with relative ease (PR 3 to 4) even with rapid divergences as severe as 2 seconds to double amplitude. In the tight-control landing task the static instabilities were not a problem and, in fact, were not even noticed by the pilots. Only for the most

unstable case, Configuration 7-3, was a problem evident and then only to Pilot B on the approach task (PR=6). It is reasonable that problems associated with the divergence should surface during the approach task in which the pilots control and attention is not as "tight".

4.5 PRELIMINARY CORRELATION WITH THE NEAL/SMITH CRITERION

Although a detailed analysis of the data from this program using the Neal/Smith closed-loop pitch attitude criterion developed in Reference 4 is beyond the scope of this report, some preliminary findings can be discussed.

The reader is referred to Reference 4 for details of the criterion; the important parameters which come out of the analysis are the closed-loop resonance and the pilot compensation at the bandwidth frequency. Recall that bandwidth can be viewed as the degree of aggressiveness with which the pilot makes changes in pitch attitude. Data for selected configurations from this preliminary analysis are presented below.

CLOSED-LOOP RESONANCE (dB)/PHASE ANGLE OF PILOT COMPENSATION (Deg)					
	BANDWIDTH (rad/sec)				
Config. No.	1.5	2.0	2.5	3.0	Average PR
2-1	3/-25	3/-6	5/20	6/42	2
2-C	-2/-25	3/-11	5/8	5/30	2½
4-1	-3/6	-3/27	-3/45	-1/59	2
6-2	-1/2	-1/27	0/48	3/61	2
2-9	3/-5	8/17	15/40	40/55	10
2-11	2/-17	7/11	11/40	19/52	8
4-2	-3/27	-2/50	3/65	8/74	6½
4-10	-2/37	4/57	12/71	32/77	9
5-1	-3/-43	0/-30	3/-18	12/-13	6
5-5	-3/-35	2/-23	9/-14	20/-8	7
6-1	0/27	6/53	16/67	30/80	10

Assuming that the application of a closed-loop pitch attitude criterion is valid - a point which must be demonstrated with more detailed analysis - the following points can be made:

- No single bandwidth yields reasonable correlation with the flying qualities boundaries of Reference 4.

- Lower bandwidths (1.5 to 2.0) are required for the satisfactory aircraft (PR < 3.5) to correlate with the 3 db Level 1 boundary of Reference 4. The boundary may indeed be different for the landing task where it is reasonable that the pilot is more tolerant of attitude oscillations than in the air-to-air tracking situation.

- Higher bandwidths (2.5 to 3.0) are required to produce closed loop pitch tracking performance consistent with the pilot ratings and comments for the unacceptable aircraft (PR > 6.5).

- Although not listed, the results for Configuration 1 indicate that there must be a limit on pilot lead capability (time constant less than about 1.0 sec for acceptable ratings, PR < 6.5) to yield reasonable correlation.

- The sensitivity of the configuration to bandwidth appears to be a very important parameter. Note that all the satisfactory configurations show small changes in closed-loop resonance while each of the unacceptable aircraft show sharp changes in closed-loop resonance as bandwidth is increased.

Obviously, this brief discussion is incomplete but it does indicate that higher bandwidths than previously estimated (Reference 6) must be used for the landing task and that the sensitivity of the aircraft to a range of bandwidths may be an important correlation parameter.

Section 5

CONCLUSIONS

The experiment described in this paper utilized the NT-33 variable stability aircraft which is capable of reproducing a wide range of aircraft and control system characteristics. Therefore, the results are largely independent of the actual aircraft employed and are restricted only by the task, range of dynamics, flight conditions and aircraft and control system parameters realized in the experiment. Conclusions which may be drawn from this experiment on the effects of control system dynamics on longitudinal approach and landing flying qualities are:

- For aircraft with significant control system dynamics, the landing task, or flare and touchdown, is the critical piloting task.

- The critical area is the last 50 feet of the landing task; landing approach flying qualities evaluations must therefore include actual touchdowns, in a realistic environment, to be valid.

- Significant control system lags create PIO's in the landing task but not in the approach task; basic aircraft problems such as low short period damping or low static stability do not create PIO's in the landing task.

- For the landing approach task (Flight Phase Category C), the longitudinal flying qualities requirements of MIL-F-8785B(ASG) and suggested revisions are not applicable to aircraft with significant control system dynamics.

- Pilot could perform the landing task with relative ease (PR 3 to 4) even with rapid longitudinal divergencies as severe as 2 seconds time to double amplitude.

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Jerry Lockenour, Northrop: The time histories you presented indicate that the pilot aggressiveness was clearly different in the landing as contrasted with the tracking task. Also, the aerial refueling task experiment in the NT-33 indicated a large change in aggressiveness, or gain, between the last five feet and the approach to the basket much like the difference you have shown between the landing and the approach. Would you compare the level of aggressiveness in the landing tasks with the tracking tasks?
Answer: Although a detailed analysis of the data has not yet been undertaken, it is clear that the last 50 feet to touchdown is where the action is. The degree of aggressiveness - the pilot gain, if you like - in this portion of the task appears to be similar to the gain used in tight up-and-away tracking tasks. In fact, it appears that the Flight Phase Category A boundaries would be more appropriate for the landing task than are the present Category C boundaries. Remember that the majority, if not all, of the existing back-up data in MIL-F-8785B for Flight Phase Category C doesn't include the critical landing task - the flare and touchdown.

In looking at the time histories for Configuration 6-1, it appears that the PIO frequency is quite different in the landing task as compared with the tracking task; therefore, the aggressiveness or gain is quite different. Not the case, observe the factor of two differences in the time scales (which I failed to point out at the time); both cases have approximately the same PIO frequency (between 4 and 4.5 rad(sec)).

Wayne Thor, ASD: The degradation of response characteristics (and pilot ability to do a task) with increasing criticality of the task... doesn't this reflect the change in pilot gain and frequency bandwidth with which the pilot attempts to perform these tasks? And therefore, couldn't we, (or has it been done) determine a human factors type limit on lag frequency and gain (separately or in combination) for a 5th through 95th percentile pilot which could than be examined over a range of expected values for a given task including the worst case combination for a given higher order system, to see if instability or some other dangerous or undesirable characteristic can be precipitated and identified?
Answer: We have been trying to model the pilot for years and I do not hold out much hope of actually measuring the pilot gain characteristics in a real task environment as opposed to a laboratory test situation. Special care is therefore required in interpreting pilot rating data for aircraft which have significant dynamic anomalies in the form of delays or lags.

Hans Stegall, NASA JSC: What is the required bandwidth for landing?
Answer: Extrapolations of the Neal-Smith criterion to the landing approach task suggested a bandwidth of 1.2 rad/sec. Although no definitive answer is available at this time since detailed analysis of the data has not been undertaken, it is clear that the appropriate

bandwidth is much higher than 1.2 rad/sec. The bandwidth is likely of the same order as suggested for up-and-away pitch tracking, about 3 rad/sec.

Of more significance than the exact bandwidth is the rate of change of resonance - tendency to oscillate - with changes in bandwidth. Good aircraft have an orderly, linear increase of resonance with increasing bandwidth. In contrast, poor aircraft (Level 3) show sudden increases of resonance as bandwidth is increased - they have flying qualities "cliffs". In my view the aircraft should be free of these sudden degradations in closed-loop performance up to 3.5 rad/sec for acceptable flying qualities (Level 2).

J.E. Buckley, McAir: Having applied the method to all of the LAHOS configurations at B.W.'s ranging from 1.0 to 5.0 rad/sec, it was found that the variation of pilot compensation with increasing B.W. provided a clearer picture of pilot opinion problem areas than any fixed bandwidth, but that the LAHOS data correlated reasonably well at the originally recommended bandwidth of 3 to 3.5 rad/sec. Pilot time delay was 0.2 sec.
Tom Twisdale, Edwards FTC: We have found that there is no deficiency that is not uncovered in tracking even if the pilot compensates.

John Schuler, Boeing: Do you think that there are other variables to be considered close to the ground?
Answer: Yes, we have really been dealing directly with the inner loop problems in this experiment. Good control of the inner loop is clearly a necessary but perhaps not sufficient condition for good approach and landing flying qualities. We would like to establish requirements or criteria to guarantee good inner loop control as a first step.

John Hodgkinson, McAir: Why do you think advanced ground simulators don't produce the PIO problems shown in the in-flight simulator?
Answer: I do not know the exact reason but there are obviously some factors missing - because of visual or motion drive problems perhaps - which cause the pilot to fly differently. In PIO prone configurations, these differences can be significant. It would certainly be useful to repeat some of the LAHOS data in an advanced ground simulator and shed some light on this anomaly. For landing tasks, at least, in-flight simulators must be included in the design and development process.

Chick Chalk, Calspan: I tried to model a simple closed-loop problem for Configuration 6-1 using acceleration at the pilot's station, pitch attitude and flight path angle. I could not get close to the PIO frequencies for flight path closures and could not get the PIO frequency with attitude without decreasing the pilot's time delay to 0.2 sec. A more complex model that includes pitch angular acceleration seems to be necessary.

THE NT-33A EQUIVALENT SYSTEM PROGRAM

(AN INFORMAL REPORT)

Rogers E. Smith
Calspan Advanced Technology Center

&

John Hodgkinson
McDonnell Aircraft Co.

ABSTRACT

This was a joint program with Calspan, McAir, Air Force Flight Dynamics Laboratory and Naval Air Development Center. As the program was only completed on 25 August 1978 the results presented here are necessarily preliminary. Results and analysis will be documented when completed.

22 September 1978

INFORMAL PAPER ON NT-33 EQUIVALENT SYSTEM PROGRAM

(VU-GRAPH COMMENTS)

Vu-Graph No.	Comments
1	The equivalent System Program (ESP) was conducted at NAS Patuxent River from 14-25 August 1978 using the AFFDL/Calspan NT-33A in-flight simulator. This research effort was intended to be exploratory and was undertaken as a joint AFFDL/Calspan/MCAIR/NADC program. The informal presentation at this meeting is an overview of the program since the data has not yet been thoroughly reviewed.
2	The purpose of the program was primarily to explore the applicability of the equivalent system technique for specification of the flying qualities of aircraft with complex, higher order, dynamics. A secondary objective was to gather additional data on the effects of pure time delays and obtain flying qualities data for lateral higher order systems since no data presently exists for this axis.
3	Self explanatory — last point introduces the next vu-graph.
4	This transfer function is representative of a modern highly augmented fighter aircraft in the power approach flight phase.
5	This vu-graph illustrates the meaning of equivalent systems as presented by MCAIR: the pitch rate transfer function amplitude and phase characteristics are matched over a range of frequencies with the simplified model shown. The numerator term can also be a variable in the matching procedure, if necessary.
6	Self explanatory.
7	Self explanatory; mismatch refers to the "cost function" which results from the matching procedure. One of the questions addressed was how large this mismatch can become before the equivalence is significantly degraded. In the cost function formulation, the weighting of amplitude and phase errors are related in a prescribed fashion; in addition, the match accuracy can be increased in one frequency range and decreased elsewhere to produce a different mismatch distribution.

Vu-Graph No.	Comments
8	Self explanatory; "pitch disturbance" refers to an intentional disturbance which was introduced into the aircraft at 20 feet above the runway to give the evaluation pilot a realistic task near the ground and prevent him from "beating the system" by backing out of the loop.
9	For this program no time delays or breakout forces were included. In general, the equivalent systems were mechanized using the NT-33 variable stability system and time delay circuit to achieve the equivalent transfer functions as shown in vu-graph No. 5. For some equivalent systems, special mechanizations were required as illustrated for Configuration P6.
	For the lateral cases, the variable stability system was used to achieve the two values of roll mode time constant and the time delay and first order lag prefilter were mechanized with special circuits.
10	Self explanatory
11	Configurations with large time delays or lags were observed to be very sensitive to inherent pilot ability (smoothness), task environment (turbulence, upsets), and pilot objectivity. The last point refers to the pilot's adherence to the task - did he try to "beat the system" by changing the task or by use of special predictive capability (fly essentially open loop). Situations were observed where the predictive pilot had the same degree of difficulty as the non-predictive pilot if he got into a position near the ground which forced him to demand immediate, accurate response from the aircraft.
	The remainder of the informal presentation was given by John Hodgkinson of MCAIR.

NT-33 EQUIVALENT SYSTEM PROGRAM

- EXPLORATORY
- AFFDL/CALSPAN/MCAIR/NADC EFFORT
- PRESENTATION - AN OVERVIEW (PROGRAM COMPLETED 25 AUG 1978)

PURPOSE

- GATHER ADDITIONAL FLYING QUALITIES DATA FOR AIRCRAFT WITH COMPLEX FCS's
  - EFFECTS OF PURE TIME DELAYS (DIGITAL SYSTEMS)
  - LATERAL DATA
- EXPLORE APPLICABILITY OF EQUIVALENT SYSTEM TECHNIQUE FOR SPECIFICATION OF F.Q. FOR COMPLEX AIRCRAFT

BACKGROUND

- AIRCRAFT WITH COMPLEX FCS'S ARE HERE, EG:
  - F-16
  - SHUTTLE
  - F-18 (YF-17)
- MIL-8785 IS NOT ADEQUATE
- EQUIVALENT SYSTEM IS A POTENTIAL APPROACH WITH WHICH F.Q.'S CAN BE EVALUATED
- TYPICAL TRANSFER FUNCTION OF MODERN COMPLEX AIRCRAFT

EQUIVALENT SYSTEM APPROACH IN NT-33 PROGRAM:

$$\frac{\dot{\theta}}{F_s} = K \frac{(s + 1/\tau_{\theta_2})\ e^{-\tau s}}{(s^2 + 2\zeta_E \omega_E s + \omega_E^{\ 2})}$$

- MATCH OF AMPLITUDE AND PHASE OVER FREQUENCY BAND (0.1 TO 10 RAD/SEC, FOR EXAMPLE)

EQUIVALENT SYSTEM PROGRAM
(ESP)

- LANDING APPROACH FLIGHT PHASE (C)
- LONGITUDINAL
  - CONCENTRATE ON EQUIVALENCE QUESTIONS
  - EXTEND EXISTING DATA BASE
- LATERAL
  - CONCENTRATE ON DATA BASE
  - QUICK LOOK AT EQUIVALENCE (DIRECTIONAL AXIS NOT A FACTOR IN SELECTED CONFIGURATIONS)
- 1 WEEK FOR EACH AXIS

LONGITUDINAL TECHNICAL QUESTIONS

- BASIC EQUIVALENCE?
- MISMATCH THRESHOLD?
- MISMATCH DISTRIBUTION?
- $L_{\alpha}\ (1/\tau_{\theta_2})$ FREE OR FIXED?
- HIGHER ORDER LAG VERSUS PURE TIME DELAY?
- OTHER LOWER ORDER SHAPES ACCEPTABLE?

  E.G. 0/2ND

LATERAL TECHNICAL QUESTIONS

- EFFECTS OF FIRST ORDER LAGS?
- EFFECTS OF PURE TIME DELAYS?
- BASIC EQUIVALENCE?

CONFIGURATION EXAMPLES

LONGITUDINAL:		ζ	ω	$1/\tau_{\theta_2}$	τ
P3:	(HOS)	--	--	--	--
P6:	(LOS)	.70	5.3	12.5	.08

- MECHANIZATION: $\frac{\dot{\theta}}{F_s} = ke^{-.08s} \frac{[.6;2.3]}{[.7;5.3]} \frac{[12.5]}{[.8]} \left\{\frac{[.8]}{[.6;2.3]}\right\}$

NT-33

LATERAL:		τ_R	τ
L3:	(HOS)	--	--
L4:	(LOS)	.44	.13

- MECHANIZATION: $\frac{p}{F_s} = k e^{-\tau s} \frac{1}{(\tau_R s+1)}$

- GENERAL $\frac{p}{F_s} = ke^{-\tau s} \frac{1}{(\tau_{LAG} s+1)} \cdot \frac{1}{(\tau_R s+1)}$

EVALUATION TASKS

- VFR LANDING APPROACH TASK
- 3 LANDINGS PER EVALUATION
  - STRAIGHT-IN
  - SMALL SIDESTEP AND HEIGHT OFFSET
  - LARGE (AGGRESSIVE) SIDESTEP AND HEIGHT OFFSET
- WIND AND TURBULENCE AS FOUND - GENERALLY NOT A FACTOR
  - PITCH DISTURBANCE
- DATA
  - PILOT RATINGS, COMMENTS
  - PERFORMANCE DATA
  - SPECIFIC QUESTIONS ON EQUIVALENCE AFTER EVALUATIONS WHEN POSSIBLE

EVALUATION SUMMARY

- LONGITUDINAL
  - 33 CONFIGURATIONS
  - 80 EVALUATIONS (3 PILOTS)
- LATERAL
  - 28 CONFIGURATIONS
  - 50 EVALUATIONS (3 PILOTS)
- 21 SORTIES/23.6 HOURS/10 DAYS

OBSERVATIONS

- SENSITIVITY OF MARGINAL AIRCRAFT (LARGE LAGS OR DELAYS) TO:
  - PILOT ABILITY
  - TASK ENVIRONMENT
  - OBJECTIVITY - ADHERENCE TO TASK
    - PREDICTIVE CAPABILITY
- PRELIMINARY DATA REVIEW

AD-A066 493 AIR FORCE FLIGHT DYNAMICS LAB WRIGHT-PATTERSON AFB OHIO F/G 1/2
PROCEEDINGS OF AFFDL FLYING QUALITIES SYMPOSIUM HELD AT WRIGHT --ETC(U)
DEC 78 G T BLACK, D J MOORHOUSE, R J WOODCOCK

UNCLASSIFIED AFFDL-TR-78-171 NL

6 OF 7

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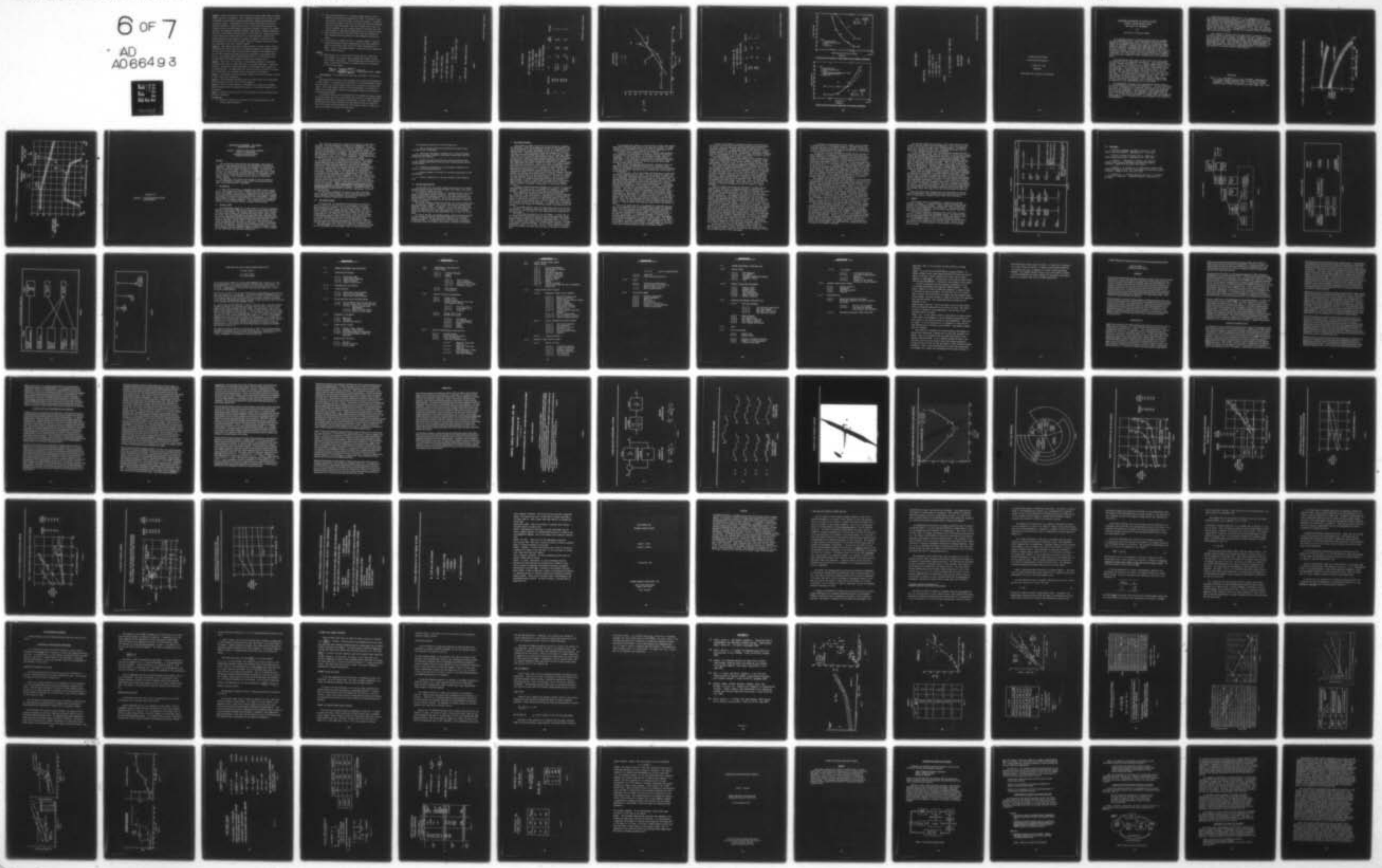
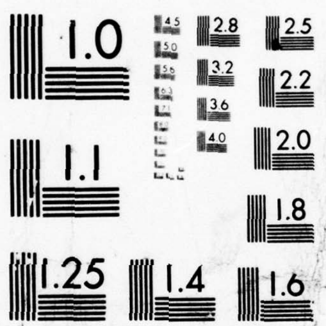




MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

Figure 1 - Analytical effort for the equivalent systems program (ESP) is planned using a variety of techniques. First, equivalent systems will be used to obtain information on equivalence, mismatch sensitivity, mismatch distribution etc., as defined earlier. Comparisons will be made with MIL-F-8785, and the K/S mismatch parameter (J. RAeS February 1976). Pilot-in-the-loop methods will include the Neal and Smith method, including an attempt to extend it to lateral dynamics, and possibly the MCPILOT program. Fast Fourier transform analysis of the pilot's input/output characteristics is also hoped for. Because post simulation checking and validation has not yet been performed, naturally this planned analytical scope cannot be complete or binding. In any case, full documentation of the data will be made. With the cautionary note that results are preliminary, some brief analysis follows.

Figure 2 - Navy fighter equivalent dynamics tended to receive similar ratings, indicating that pilots might not be particularly sensitive to mismatch between the high and low order systems (HOS and LOS). However, pilot comments have not yet been reviewed and clearly these are important in establishing equivalence. The slightly worse ratings obtained for the "L_α free" matches may be significant, but equally might be within normal rating scatter.

Figure 3 - The effect of pure delay in longitudinal dynamics is reasonably consistent with trends of added lags seen in the LAHOS experiment. These data indicate a threshold in rating sensitivity to delay which did not appear in the LAHOS results. The LAHOS high order system re-flown in the ESP (the square point marked HOS) has a somewhat better rating than its equivalent.

Figure 4 - Navy fighter lateral equivalent systems received very similar ratings to their high order counterparts.

Figure 5 - This remarkably consistent data set clearly shows the rating degradation produced by control lag in lateral dynamics.

Figure 6 - Pure time delay produces a degradation in rating with a somewhat larger threshold than for longitudinal dynamics. The eventual degradation is pronounced.

Figure 7 - Within the main purpose of the experiment, several peripheral questions were addressed.

Longitudinal:

(1) Equivalent systems were obtained for configurations flown in the earlier LAHOS experiment.

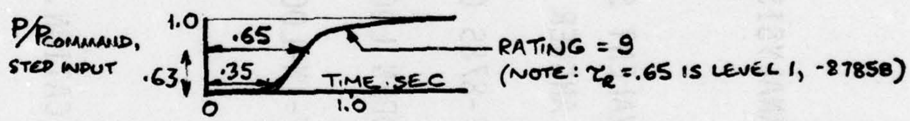
(2) The series feel system (i.e., position command concept) used in LAHOS can be analytically represented by about .05 seconds of equivalent delay in aircraft response to pilot force inputs. A parallel system (force commands) was used in the ESP, and a few runs were made with .05 seconds delay added. This will provide data on force vs. position commands, and will indicate whether a relatively fast feel system produces a degradation commensurate with its equivalent delay.

(3) Some variations in gain were made because there were indications that ratings become more sensitive to gain changes when delays are large.

(4) A short series of flights was made to determine whether a lead-lag network cancelling the phase lag generated at the short period frequency by time delay, would cancel the rating degradation due to delay. In fact, the rating worsened. Lead-lag at higher frequencies did not improve matters.

<u>Lateral:</u>

(1) Measuring the time to reach 63% of steady state for a high order step response is one way to check acceptability against roll mode time constant requirements. The data shows that configurations with delays or lags cannot reliably be evaluated this way, as indicated in the following sketch.



<u>Final Remarks</u> - It is believed that the major objectives of the ESP will be met.

The exploratory structure of the experiment also allowed quick examination of minor issues. It is expected that in addition to the subexperiments already noted, the data will yield valuable insight into the effects of piloting techniques. Also, some comments indicate pilot discrimination of very high frequency gain characteristics, which is an unexpected finding of significance to control system designers.

The dramatic lateral control problems and pilot induced oscillations caused by lags and delays show that a comprehensive set of data for lateral-directional augmented effects, both in landing and up-and-away, is needed. Data comparing ground-based and in-flight simulators are also needed so that the respective roles of the analytical and experimental tools currently used for flight control system design are brought into perspective.

PLANNED ANALYSIS FOR EQUIV. SYSTEM SIMULATION

- o EQUIVALENT SYSTEMS -
  ANSWER BASIC QUESTIONS
- o MIL-F-8785 COMPARISONS
- o K/S OPEN LOOP CORRELATIONS
- o PILOT-IN-LOOP: o NEAL & SMITH
  o MCPILOT (PAPER PILOT)
- o FFT
- o PUBLICATION. PRELIMINARY ANALYSIS . . .

FIGURE 1

LONGITUDINAL

NAVY FIGHTER RESULTS ON

- o BASIC EQUIVALENCE
- o L_α FIX - FREE
- o MISMATCH THRESHOLD
- o TIME DELAY EQUIVALENCE

CONFIG	HOS/LOS	ζ/ω	L_α	ξ	MIS MATCH	RATING
	HOS					2
1	LOS	1.0/1.4	.5	.14	40	2
	LOS	.6/3.5	6.3	.09	20	3
	HOS					3
2	LOS	1.5/1.7	.54	.14	20	3
	LOS	.7/5.3	12.0	.08	.7	4

FIGURE 2

MCDONNELL AIRCRAFT COMPANY

LONGITUDINAL

DELAY EFFECTS

$\zeta = .57$

$\omega = 2.3$

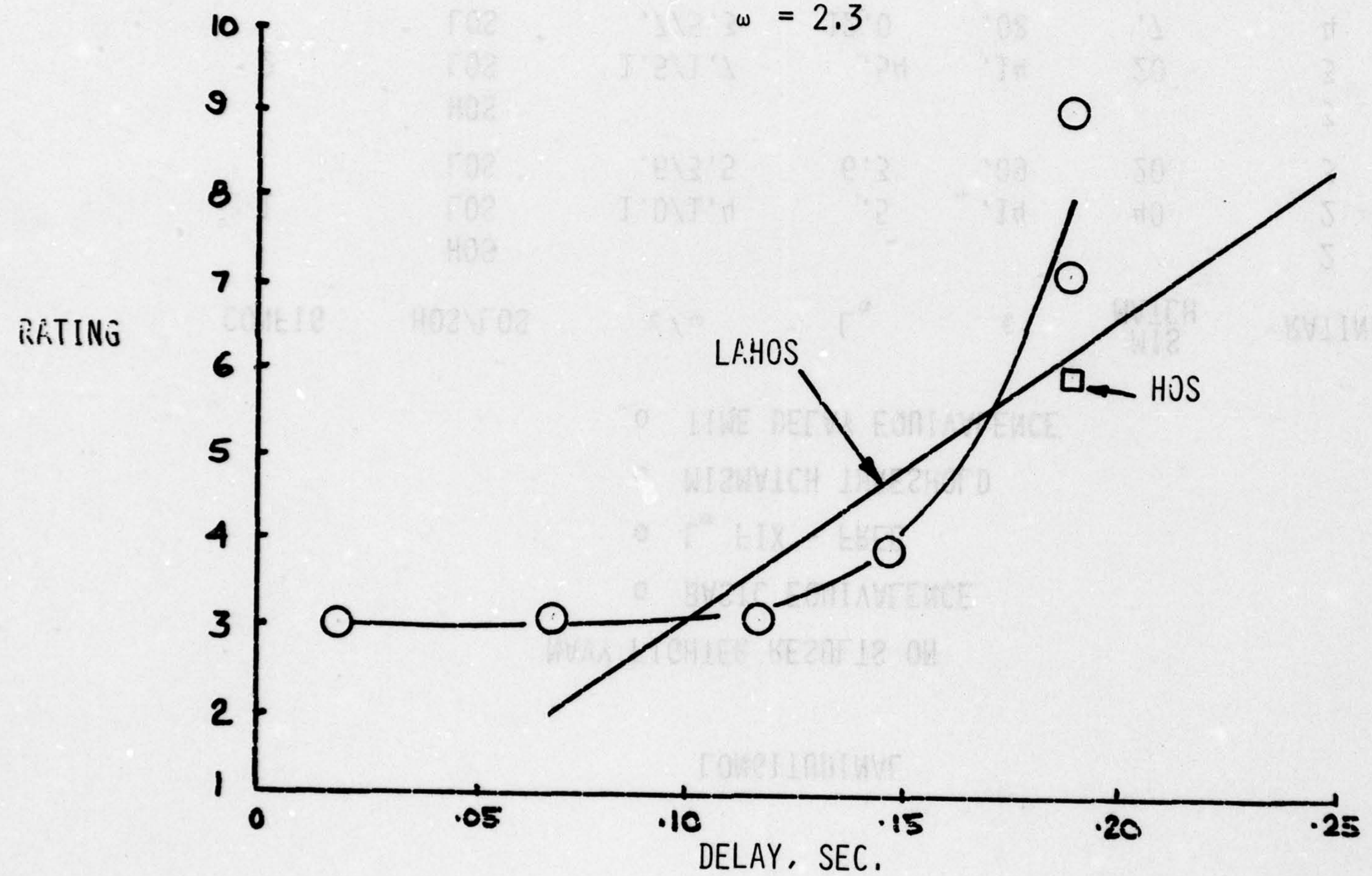


FIGURE 3

MCDONNELL AIRCRAFT COMPANY

LATERAL

NAVY FIGHTER RESULTS ON:

- o BASIC EQUIVALENCE
- o TIME DELAY EQUIVALENCE

CONFIG	HOS/LOS	τ_R	DELAY	RATING
1	HOS			4
	LOS	.44	.15	3
2	HOS			4
	LOS	.37	.13	4, 3

FIGURE 4

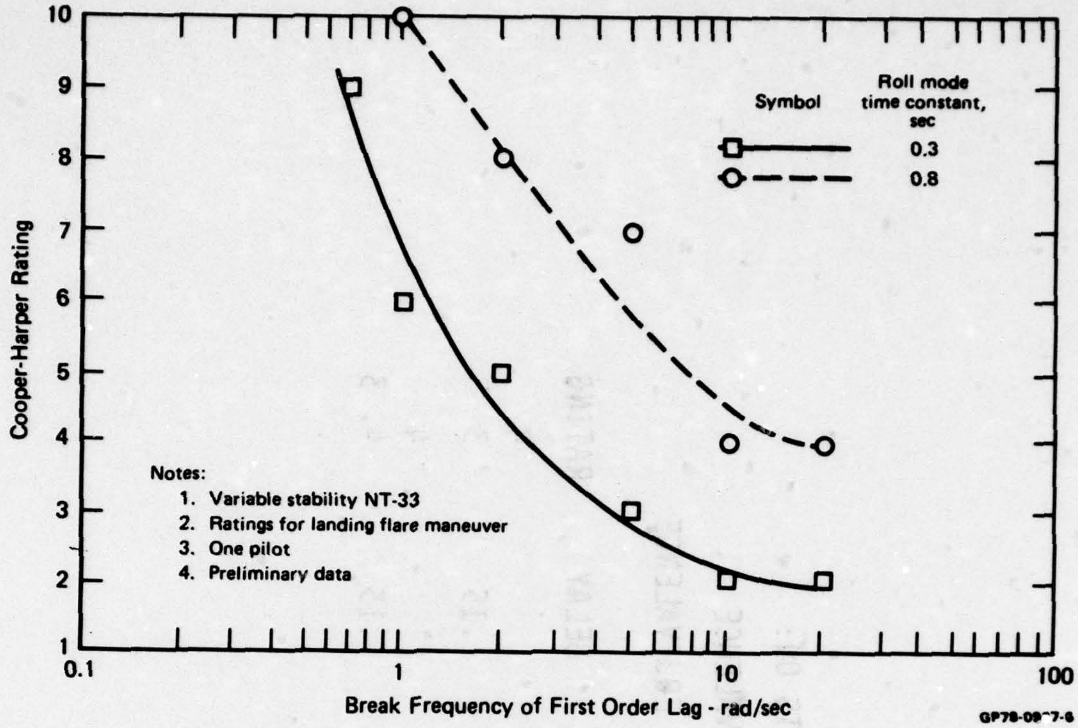


FIGURE 5

EFFECT ON PILOT RATING OF FIRST ORDER LAG IN LATERAL RESPONSE

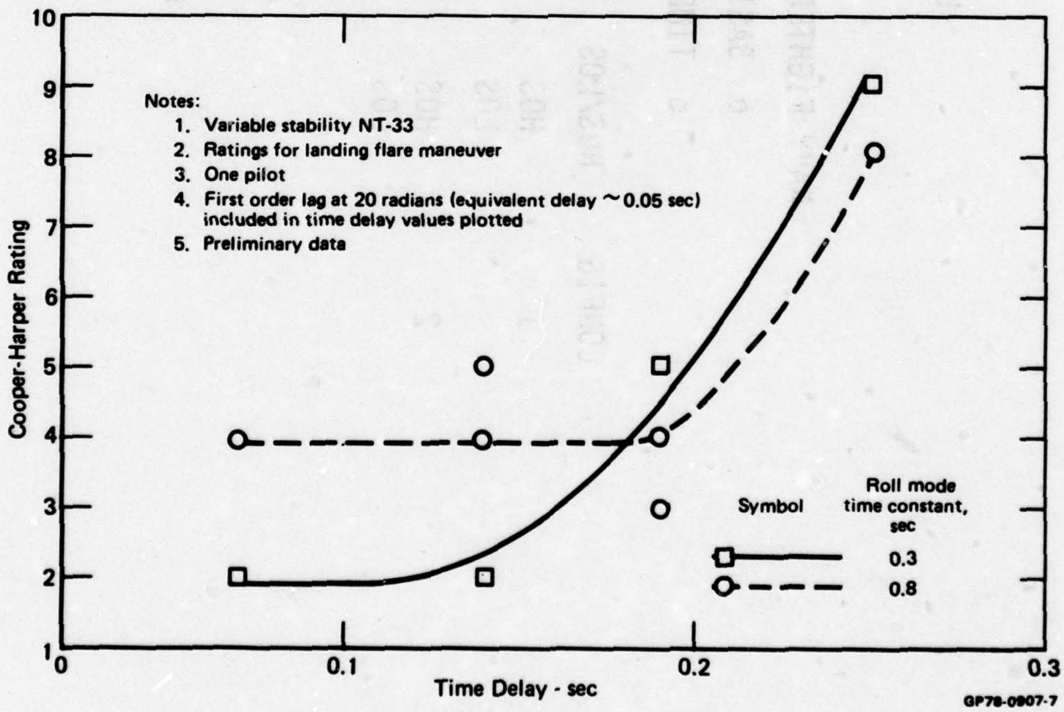


FIGURE 6

EFFECT ON PILOT RATING OF TIME DELAY IN LATERAL RESPONSE

SUBEXPERIMENTS

LONGITUDINAL

- o EQUIV. SYSTEMS FOR LAHOS
- o FEEL SYSTEM $\stackrel{?}{=}$ DELAY
- o GAIN EFFECTS
- o LEAD COMP. OF DELAYS

LATERAL

- o .63 (TIME RESPONSE) $\stackrel{?}{=}$ TIME CONSTANT

EXPLORATORY

PRELIMINARY

FIGURE 7

MCDONNELL AIRCRAFT COMPANY

Shuttle Orbiter Flight

Control System Evaluation

L. Brown & M. Moul

NASA/LaRC

[This paper not received for inclusion]

NASA/DFRC EVALUATION OF DIGITAL FLIGHT CONTROL SYSTEM DELAYS USING THE DFCS F-8 AIRPLANE

by

Don Berry and Bruce Powers

One problem unique to digital flight control systems is the introduction of a pure transport delay due to the sampling and computation cycle. In order to provide information to validate criteria that might be useful for these types of systems, a program was conducted on the DFRC F-8 digital flybywire airplane where the transport lag could be easily varied. Two landing tasks were evaluated: tightly controlled landings where the pilot attempted to touchdown accurately at a given spot; and loosely controlled landings where the pilot accepted large variations in the touchdown point.

The approaches were made at 180 knots with idle power to simulate low L/D approach conditions. The longitudinal control system consisted of a C* command augmentation system. However, it was not optimized for these flight conditions and provided only nominally satisfactory longitudinal handling qualities. Pure transport lag ranging from 0 to 100 milliseconds was added between the pilot and the control system. The results are shown in figure 1 for the two landing tasks. The longitudinal ratings for the loose control task degrade slightly with increasing delay time. The ratings for the more difficult tight control landing task show a significant degradation with increasing delay time in addition to the degradation due to the task.

One promising criterion for correlating these results is the Neal-Smith criterion of reference 1. In this criterion, the pilot is modeled as a lead-lag network and the pilot gain and phase are adjusted to provide good closed-loop response over a given bandwidth. The unknown in the formulation is the relationship between the bandwidth requirement and the task. It is certainly reasonable to assume that the bandwidth requirements will increase as the task becomes more difficult and demanding.

The pilot ratings from figure 1 are compared to the Neal-Smith criterion in figure 2. Two bandwidths were selected which provide a reasonable correlation with the criterion. For the loosely controlled landing task, a bandwidth of 2.5 rad/sec was used and the criterion shows a good correlation with the pilot rating as a function of transport delay. For the tightly controlled landing task, a bandwidth of 3.0 rad/sec was selected. Again, the trend of pilot rating with transport delay correlates well with the criterion.

The results indicate that the Neal-Smith criterion can provide a means of analyzing the effects of pure transport delay in the flight control system. However, further research is needed to establish the relationship between the bandwidth requirements and the task. For situations where the task/bandwidth relatiohnship is know (such as with an existing flight vehicle and task) the method can provide a reasonable means of predicting the effects of control system modifications.

Reference

1. Neal, T. P.; and Smith, R. E.: An In-Flight Investigation to Develop Control System Design Criteria for Fighter Airplanes. AFFDL-TR-70-74, vol. 1, Air Force Flight Dynamics Lab, Wright Patterson Air Force Base, Dec. 1970.

FIGURE 1 _ EFFECT OF ADDED TRANSPORT LAG ON F-8 DFBW CONTROL

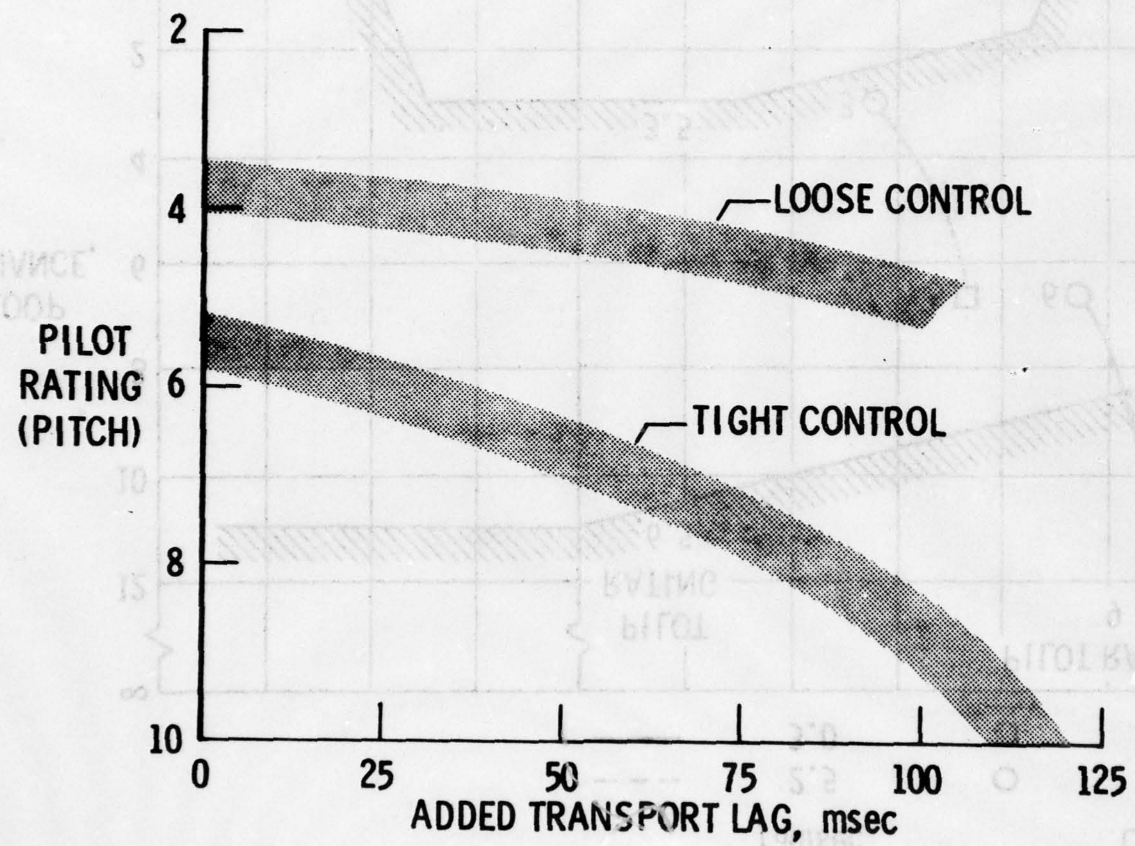
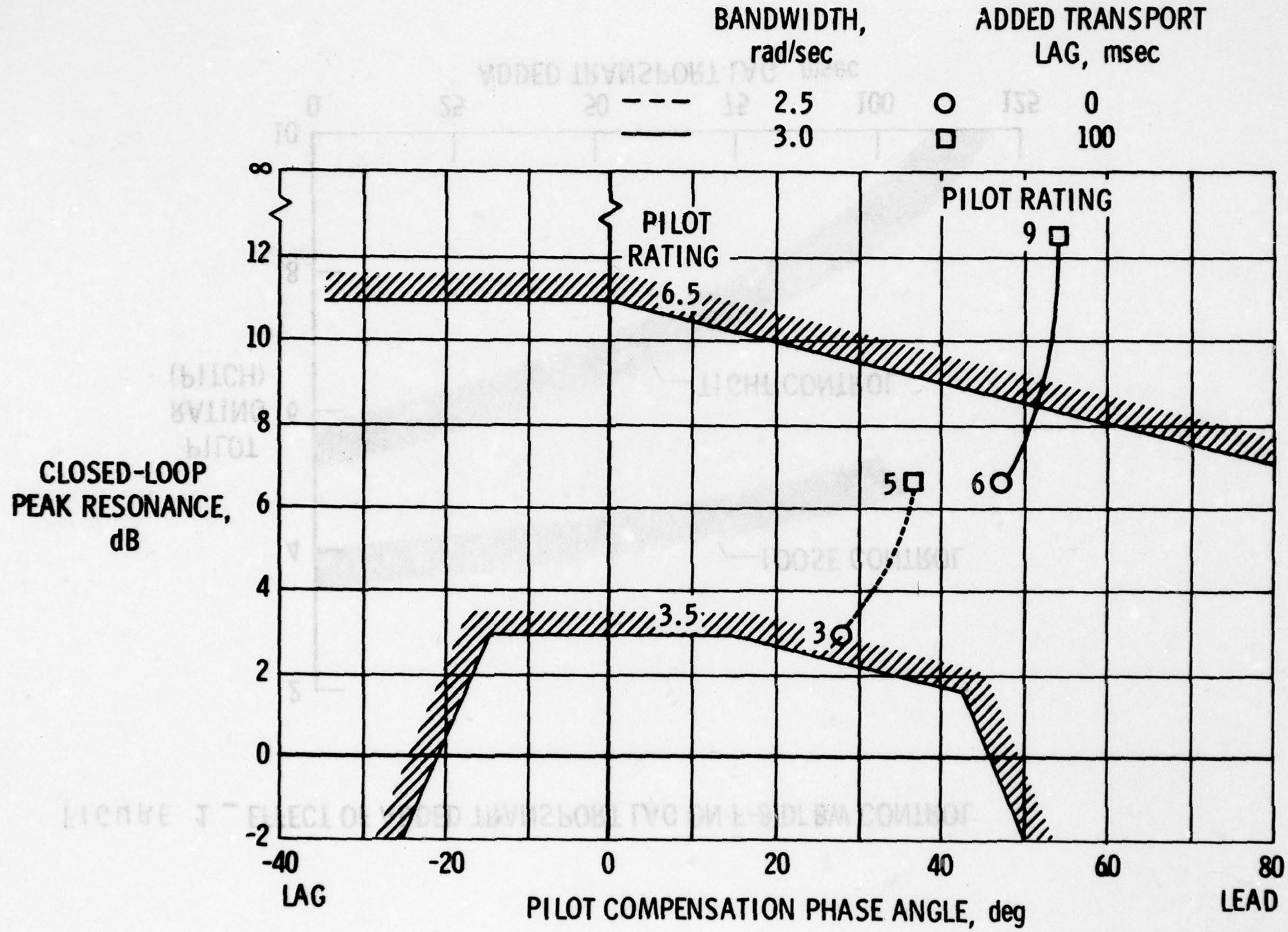


FIGURE 2 _ F-8 DFBW RESULTS AND NEAL/SMITH CRITERION



SECTION VIII

SESSION 5: FUTURE DIRECTIONS AND THE MIL-PRIME-STD

AERONAUTICAL PROCUREMENT - THE PRIMARY SPECIFICATION SYSTEM

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Deputy for Engineering
Aeronautical Systems Division
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Abstract

The Aeronautical System Division has developed a new system of writing and using specifications for the Procurement of Aeronautical Equipment. This new system requires the statement of requirements in terms of operational needs with values left blank. The values are filled in for each program. In addition, a handbook will be provided for each specification that provides rationale for requirements, guidance for use of the specification and will act as a depository for lessons learned. These documents will be written to cover board product families.

The paper will also address the relation of this new system to Office of Management & Budget circular A-109 which requires mission needs rather than technical requirements.

I. Introduction

In April 1976 the Office of Management and Budget issued circular A-109. This document will have a profound effect on the way we budget and procure systems in the future. During this same time frame the Air Force's Aeronautical Systems Division undertook a new program to upgrade its methods of writing and using specifications and standards. Together these changes are going to have an effect on procurement of Aeronautical Systems and the Aerospace community. This paragraph briefly addresses circular A-109 and provides details on primary specifications.

II. Circular A-109

The Congressional Budget Act of 1974 (Public Law 93-344) establishes reforms in the overall budget and procurement cycle and becomes effective with the FY 1979 budget. The circular basically requires that a request for funds should be based on a mission need rather than a specific hardware item. This was established to encourage innovation and competition in creating, exploring, and developing alternative system design concepts. Figure 1 illustrates the difference between the current and the A-109 (mission) approach. Figure 2 views a typical Air Force Request, but rather than specifying a solution, we now accept a variety of approaches and select the best one. The circular specifically states that a "need should not be defined in equipment terms, but should be defined in terms of the mission, purpose, capability, agency components involved, schedule and cost objectives, and operating constraints."

This new policy results in a conflict with usage of current specifications. The specifications of today are, in themselves, not bad documents as many have claimed, but rather are misapplied. In reality, today's specifications are written to procure a specific product. Over the years the philosophy of the detail specification has spread into research and development procurements. The detail specification in the R&D environment is one of the largest contributors of problems within the system yet in most cases it is not recognized. Recent efforts have recognized this and led to efforts to correct the problems.

In essence most of the problems associated with these documents was the application of the specification and standards. The Defense Science Board Task Force reviewing this area initial findings were reported by Deputy Secretary Clements to the Military Department Secretaries: "The Task Force has concluded the content of specifications and standards is not the primary contributor to unnecessary contract costs although there is a continuing need for evolutionary improvement. The main cause of cost escalation was identified to be in the application, interpretation, demonstration of compliance and enforcement of specifications and standards in RFPs (sic: Request for Proposals) and contracts. This, therefore, is a fertile area for effective cost reductions in the acquisition process." Deputy Secretary Clements directed the Department Secretaries to "Institute Procedures and Policies to control blanket contractual imposition of such specifications and standards. <u>These controls should be structured to force technical activities to tailor requirements to the essential specific operational needs of the end item or system.</u>" (Underlines added) The overall theme is to force tailoring and state requirements in terms of operational needs.

The Task Force recommended two steps in its April 1977 final report; an evolutionary program to improve existing specifications, and an immediate program throughout the services and industry to improve the climate of applying the specifications.

III. <u>The Primary System</u>

In January 1976, the Aeronautical Systems Division started an effort to review its own utilization and development of military specifications and standards. At the time, there was no attempt to marry the specification system and A-109 budget system. What has evolved, however, is a systems approach to specifications in the R&D cycle that fits the objectives of A-109 and meets the long range recommendation of the Defense Science Board. While the A-109 looks toward mission approach, these specifications view mission needs and at the same time have evolved into a new document that forces tailoring. It is interesting to note in the experimental usage of this new system the tailoring philosophy has carried over from specifications into other items such as data items and management plans.

The ASD approach was to develop a specification system that viewed both the development and reprocurement cycles of the specifications.

The essential objectives of the ASD program are:

a. Have a structure which will facilitate and "force" selective application (tailoring).

b. Insure that requirement statements are in terms of system/equivalent performance related to operational needs rather than dictating specific solutions.

c. Provide an overall specification system that marries the current military specification and the type of specification defined by MIL-STD-490.

d. Provide the rationale for the requirement statements contained in specifications/standards.

e. Provide guidance to the user for selective application of the requirements.

f. Provide a depository for lessons learned in each technical area.

IV. The New Specifications

Are they new? No! But rather taking the best parts of a variety of specification systems from both government and industry and putting them into logical, single structure.

In developing new weapon systems, our military services are facing sharply rising costs from two directions. The first is due to the highly sophisticated equipment required on the modern battlefield and the maintenance of that equipment. The second cost driver is not very visible; it's the way we write and use our specification in a world of advancing technology.

We live in a rapidly changing environment and we need documentation that is adaptable to dhange. Our current specification systems is not adaptable to rapid change. Consequently, it has come under repetitive attack as a major cost driver. However, every group investigating the system has generally come to the same conclusion. It is a required system, but fine tuning and slow improvements will not meet our future needs.

In various aircraft prototyping programs, ASD has found giving contractors maximum flexibility and minimum supervision has resulted in technical successes. We must apply this flexibility to all weapon systems programs. To accomplish this, we have developed a new series of documents called the Primary Specification, Standard, and Handbook.

V. The Primary Concepts

This concept is aimed at new types of specifications, standards, and handbooks. These new documents are written for the contractor and for the government to provide guidance and set the framework to build specifications for product development as well as for the actual procurement cycle. The objectives of these new documents are to "force tailoring" and specify operational needs, and at the same time, improve other facets of the military specification program. This system will consist of three types of primary documents; specifications, standards, and handbooks. The primary specification (Mil-Prime) are aimed at specifying operational needs and general parameters for a physical product family with the specific values left blank. The primary standard (Mil-Prime-Std) provides the criteria and qualities applicable to a physical product but is not used to procure any actual product, and in many cases is very similar to our standards of today. The primary handbook (Mil-Prime-Handbook) contains technical rationale for the requirements stated in each primary specification and standard, provides guidance for applying the specifications and standards, and is a depository for lessons learned in each technical area. We would not develop a primary document for each individual product or service; this would just duplicate the 44,000 military specifications and standards in today's system. Instead, a broad family grouping will be established. These Mil-Prime product families will cover such areas as airborne radios, fasteners, parachutes, landing gear, or the entire aircraft structure.

The prime standards basically encompass the "-ilities" (reliability, maintainability, etc) as well as "standard test methods" or "climatic extremes." The handbooks will explain where a requirement came from and why we require it for both the specifications and standards.

These "Mil-Prime" documents, with the operational values blank will force tailoring. They also provide a depository for "corporate memory" and will allow others to question our requirements based on facts. No longer can anyone say "you must do it this way because the specification says so," and so the argument ends. Now the argument must rotate about the rationale contained in the handbook ... a logical point of departure. As new lessons are learned and technology changes, the handbooks will be updated.

As the specification is used, the blanks are filled in by the project engineer. The handbook assists in this filling-in process by showing how to fill in the blanks and, most importantly, the rationale for the requirements.

Of course, these documents are not just blanks. A closer look at the primary specification would show not only blank operational needs, but in addition, a section on interface would be provided. The project engineer would provide detail interface requirements such as a size of a bay, a coupler's part number, even available power and type, or details on government furnished equipment. In the technical community, we have standardized values, such as safety factors. These would be maintained within the document, but again with the back-up rationale in the handbook.

In developing this concept, it was found that in some cases requirements within specifications could not be logically verified. In prime documents, any requirement must be verified by a corresponding verification, whether it be a test, analysis, or inspection.

Two other typical problems are trying to find the right person to ask a question on a specification; and second, finding the individual accountable for these documents. In the present system, they are "anonymous." A new facet will be the inclusion of name, address, and phone number of the individual responsible for the document, keeping it up to date, in a manner similar to the crew chief who has his name printed on the side of an aircraft.

While we have described the specifications, standards follow the same basic format; the difference being that the "standard values" appear in the document and that we are not procuring an item of hardware. In writing the primary specification, we envision only the standards as being referenced.

In practice (sic: Fig 3) the primary specification is a guide that the project engineer must now tailor. He must examine the blanks, fill in the applicable operational needs based on user input, provide interface requirements, and tailor the test section to the mission needs. The document has been designated a Type I specification, and it is a development specification stating the performance requirement for design or engineering development in terms of the operational needs. The Type I specification becomes part of the Request for Proposal. The government evaluates the contractors' proposals against this document and issues a contract. The item is them developed and after it has been found acceptable, either the contractor or government would prepare a Type II specification. This is a product specification stating detail design requirements for procurement of a product in terms of specific design needs, basically what today is a military specification. It may even be possible that from one Type I specification a series of Type IIs would evolve, such as for new components that were developed as part of this effort. Today, too often we directly write the military specification (Type II). This is almost like having the answer before the question is asked.

This is not really new but is an adoption of a technique that has been very successfully used in development/reprocurement of component items, such as microcircuits and switches within today's specification system. For some component series, a general design specification exists and in a manner they contain blanks. The blanks are stated by the term "as specified." A development document is provided by the procuring agency which states the "as specifieds." After development, a final detail specification is issued for reprocurement. The primary specification is similar to the general design document and the Type II is the same as the detail specification. One improvement is the Type I specification, which now would provide a standard bridge from the start to the final document missing in today's procurement method.

At first, this appears to increase the number of specifications and it may if we were to view the military specification system only. In reality, there are additional specification systems within the Department of Defense. In many cases today, rather than use a specification, an exhibit of MIL-STD-490 specification is prepared. A major problem exists in the 490 system. In many cases, the hardware documentation that is developed is not available to outside activities; and as a result, standardization efforts take a second seat.

With the development of a prime document in each technical area, exhibits or MIL-STD-490 documents will no longer be required. Figure 4 shows the relationship of the current specification systems to the prime system. This will result in using one baseline; the prime specifications rather than the various documentation systems in use today.

One thing this program accomplishes is to raise the level of the specification to a higher level in the procurement tree. We have been too concerned with nuts and bolts and individual items such as particular radios and radar units. The Primary Concept allows us to move up to entire functional areas such as an aircraft structure, offensive avionics, or even to categories such as automatic test equipment and simulators. Within the Department of Defense, this method has been very successfully utilized at the component level. For higher order procurements, this same concept can be used. These are the areas the Air Force program is attacking. With experience, it may even be possible to view whole systems such as the air vehicle or support equipment.

This new concept also offers the Department of Defense with an opportunity to develop a new numbering system for specifications. Today's system is based on allocation of numbers to branches and assignment thereafter in numerical order. This does not facilitate data retrieval or provide a convenient method method to trace down products to avoid duplication. One suggested approach has been that each approach has been that each primary specification area be assigned a number. The handbook would have the same number. The Type I and Type II specifications would be issued an additional number to the basic prime number such as a dash number. In areas where the primary document covers a broad technology, additional coding may be necessary. The end objective is to provide a numbering system that would allow easy examination of all products already available within DOD that have been developed within any one class. When we combine this new numbering system with all types of specifications in one system, we also move forward in achieving another important goal standardization. This is accomplished by putting at the designers' fingertips what is available in a logical order not only what is in the current system, but also the MIL-STD-490 specification that today would have been left out. For example, let us assume we want to use a blind rivet on an aircraft. The primary area would be fasteners, and we would have additional coding for bolts, nuts, screws, rivets, etc. The end number is hown in Figure 5.

On the whole, the number of development specifications will decrease and the number of detail specification will remain the same. But we <u>will</u> know where things are, and what has been developed with DOD. This in itself will be a vast improvement.

In viewing the standardization we must remember that the amount of standardization is a management decision. The specification and standards only serve as a catalog of what is available. Utilization of the new numbering system will improve standardization by providing a better picture of what is available.

At this point, we have discussed the Primary Concept and what it does accomplish. The movement ot operational needs will allow different contractors to bid different solutions to a particular problem rather than solving the mechanics of single solution as described in so many of today's specifications. The end result will be a better product more responsibe to the user's needs. It will encourage innovative design and, at the same time, reduce gold plating, because so many of the "cover your ______" (filling the blank) documents will no longer be thrown on the stack of specifications. Another point in this area is the forced tailoring which will contribute very strongly to meeting these objectives. The concept of broad primary documents will have an effect of reducing tiering or references from one document to another. This will contribute cost savings in the long term as well as a reduction in paper work. It also removes problems where one referenced document conflicts with another referenced document.

One document that has lingered in the background of this discussion is the handbook. Not much has been said other than it will provide the rationale and instructions for the specifications and standards. If we examine this new document carefully, we can find that it is one of the most most important aspects of this program. Today, we do not have a true depository in each technical area to retain our lessons learned. These handbooks would fulfill this need. In viewing the rationale behind requirements and criteria, these may change when conditions or state-of-the art progresses. Today, changes to criteria can be very difficult to effect when you do not know why a factor was established 30 years ago. It will also open up our rationale to the public for our requirements, adding an important check and balance that is missing today. It will allow us to apply the lessons learned from one weapon system to another since we are using the "same specification." Only the blanks change and give us the same baseline for different programs.

How does the government engineer fit in? First, for each primary document, there will be known focal points. We project him to be a busy man maintaining the handbook for his specification or standard. In the real world, we foresee very little change to the specification or standard; but constant change to the handbook. The engineer with the procurement team will be required to fill in the blanks and prepare the Type I specification. This is a prelude to an extremely hard task, that of source selection. Determining if a proposed solution can be accomplished as stated and its potential for success will not be a simple task and will require the highest degree of professionalism on the part of the government, the operational test program will provide the engineer with another challenge.

How does this affect the contractor? He now has latitude to truly design a solution within his area of expertise. But on the other side of the coin, he must now take a total systems approach. The handbook gives him a baseline to depart from. In meeting the needs of the Air Force, his responsibility has vastly increased.

Is this concept a dream? No! The Aeronautical Systems Division's (ASD) Deputy for Engineering, under the direction of the Air Force Systems Command and Headquarters, United States Air Force, is already moving into this program with an impressive set of primary specifications being prepared. These include electronic countermeasures equipment, landing gear, parachutes, and airframe structures. Some of these documents are going to take almost two years to develop. This is not an easy task. When the program will be finished in 1981-82 time frame, ASD will have less than 100 documents. The Department of Defense (DOD) is viewing this effort to determine if it can be applied to all DOD agencies. Success at ASD may result in change throughout the entire military establishment.

While these new documents are being prepared, the overall philosophy is being used in development of current system specifications. The F-16 and YC-14/YC-15 aircraft are using this concept in their procurement. Further, many individual equipment items will be procured in this manner. Initial findings in use of this concept to the CX-1 program (YC-14/YC-15) show significant improvements in its use. In the case of this program, both contractors have found savings in being able to respond to an operational need. For example, the RFP does not have a fastener specification, but rather views the aircraft mission. Both contractors have approached the fastener design from a different standpoint; and in both cases, the cost savings were in the range of the cost of one aircraft in this program.

Both contractors have indicated that the procurement and specification methods used in this program have saved $50-75 million (actual data not releasable as program is still under source selection).

VI. Summary

Over the past years, the Department of Defense has tried many different techniques. It seems that many of these techniques have not resolved our procurement problems. A major contributor to these problems has been the application of specifications. The specifications have been a common denominator over the years and possibly these documents have been the problem.

The Mil-Prime program in combination with A-109 offers an opportunity to provide a meaningful procurement system. We have also found the tailoring philosophy of the Mil-Prime concepts has flowed to data items and management plans in their use on a particular program.

The Mil-Prime Program is new, and ASD's testing of the new system will result in fine tuning over the next few years. Results so far indicate this is a viable approach to vastly improve our procurement of new systems.

DOD ILLUSTRATION

USING CURRENT APPROACH			USING MISSION APPROACH	
	1977	1978		
BUDGET ACTIVITY:	AIRCRAFT AND RELATED EQUIPMENT	TACTICAL PROGRAMS	MISSION:	TACTICAL AIR WARFARE
BUDGET SUBACTIVITY:	ADVANCED TACTICAL FIGHTER	COMBAT AIRCRAFT TECHNOLOGY	MISSION AREA:	INTERDICTION
BUDGET CATEGORY:	ADVANCED DEVELOPMENT	ADVANCED DEVELOPMENT	MISSION NEED:	*ALL WEATHER STRIKE CAPABILITY
R & D PROJECT(S):	ADVANCED TACTICAL FIGHTER	COMBAT AIRCRAFT TECHNOLOGY	R & D PROJECT(S):	1. ALTERNATIVE MANNED FIGHTER DESIGNS 2. REMOTE PILOTLESS VEHICLES 3. MISSILES 4. OTHER COMPETING CANDIDATES
FUNDS REQUESTED:	$ 1 MILLION	$ 2.5 MILLION	FUNDS REQUESTED:	$____

*BUDGET NARRATIVE REFERS TO MISSION NEED BUT IS BIASED TOWARD MANNED AIRCRAFT SOLUTION.

FIGURE 1

VII. References

1. Office of Management and Budget, Circular No. A-109, "Major System Acquisitions." Washington, D.C. 1976, 12pp.

2. Office of Federal Procurement Policy, Pamphlet No. 1, "Major System Acquisitions." Washington, D.C. 1976, 36pp.

3. Carlyle, L. "Comparison of Military and Commercial Design-to-Cost Aircraft Procurement and Operational Support Practices." AFFDL-TR-75-147, April 1976, 125pp.

4. Quigley, D. L. and Reel, R. E., "Comparison of Military and Commercial Design-to-Cost Aircraft Procurement and Operational Support Practices." AFFDL-TR-75-64, July 1975, 213pp.

5. Borklund, C. W., "Mission Budgeting and A-109: Procurement is Headed for a Shape-up." Government Executive, September 1977, pp. 12-21.

A MISSION APPROACH

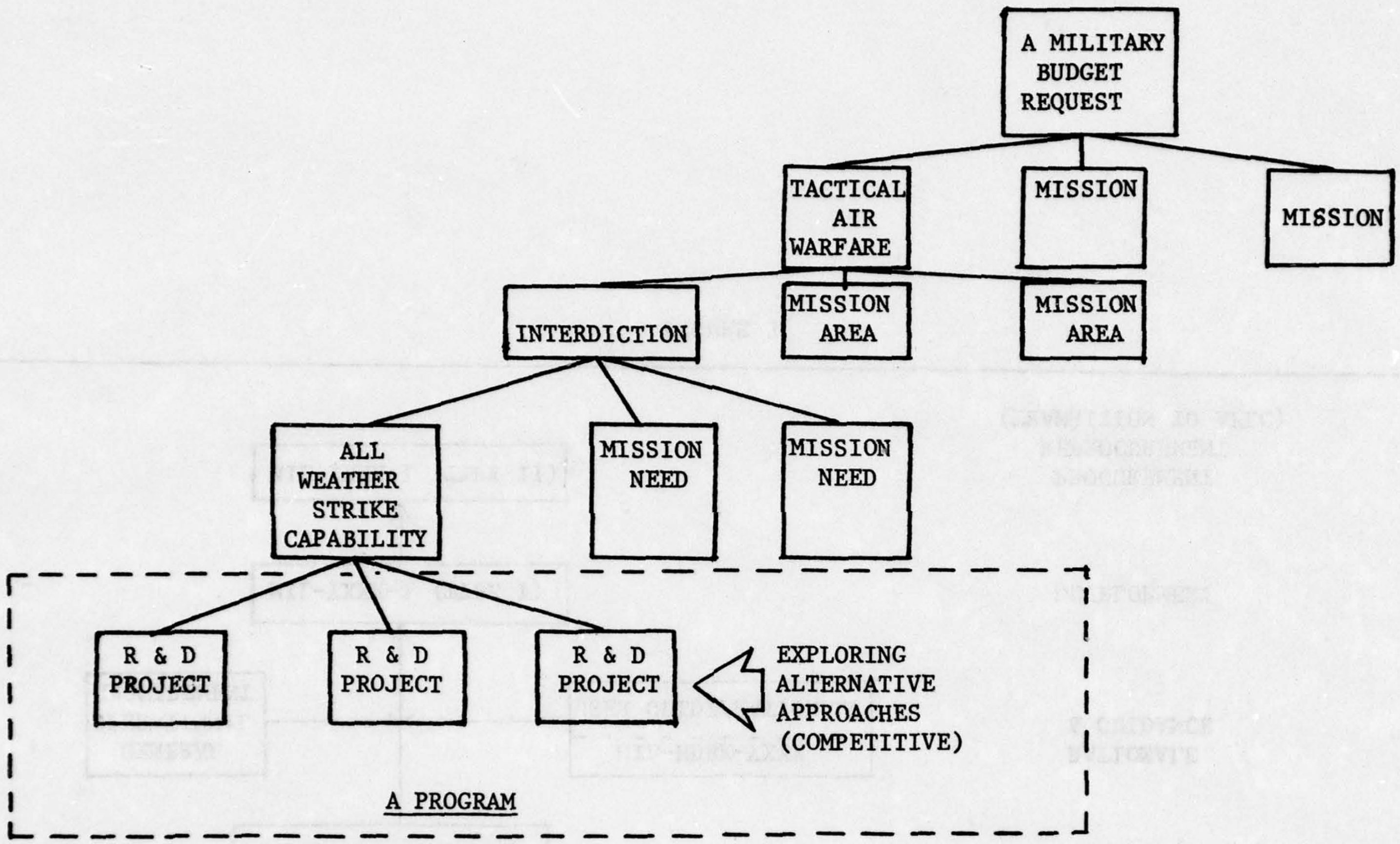

FIGURE 2

DOCUMENTATION FORMAT

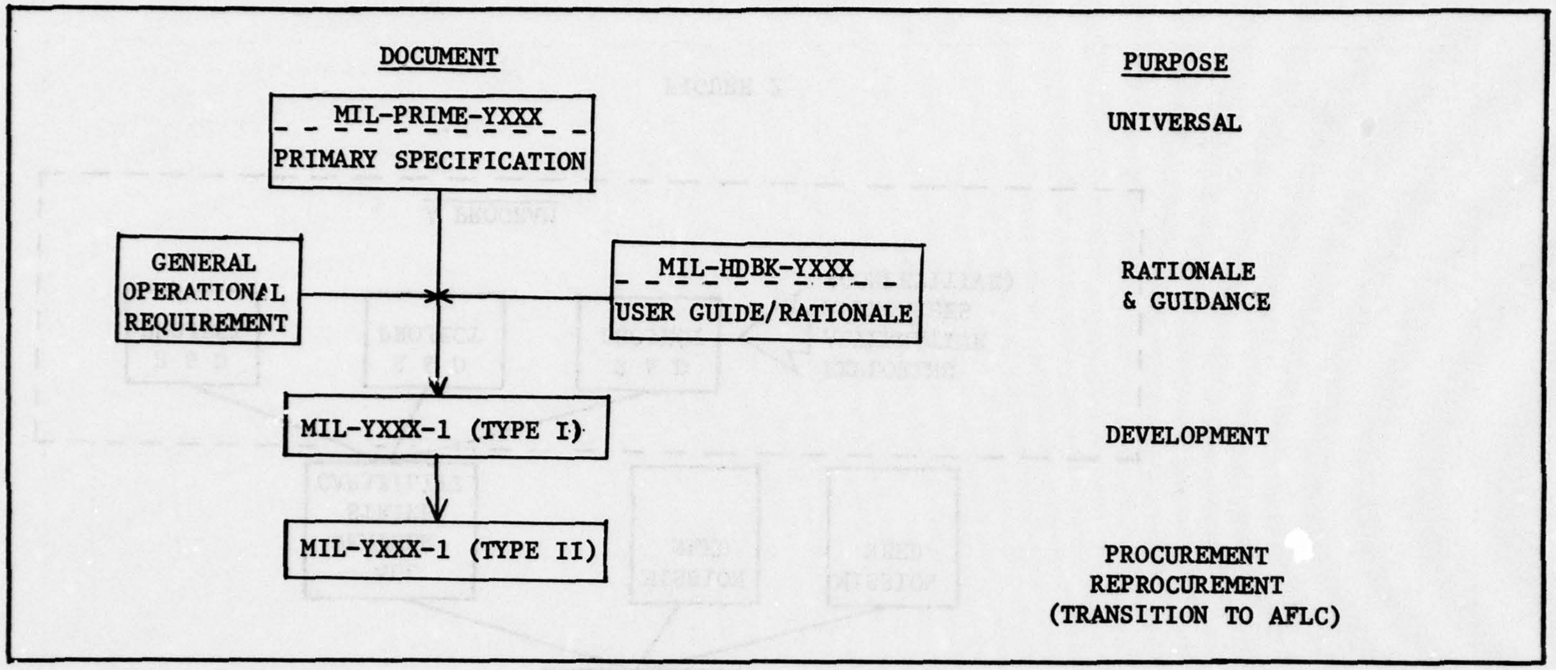

FIGURE 3

SPECIFICATION/STANDARD PERSPECTIVE

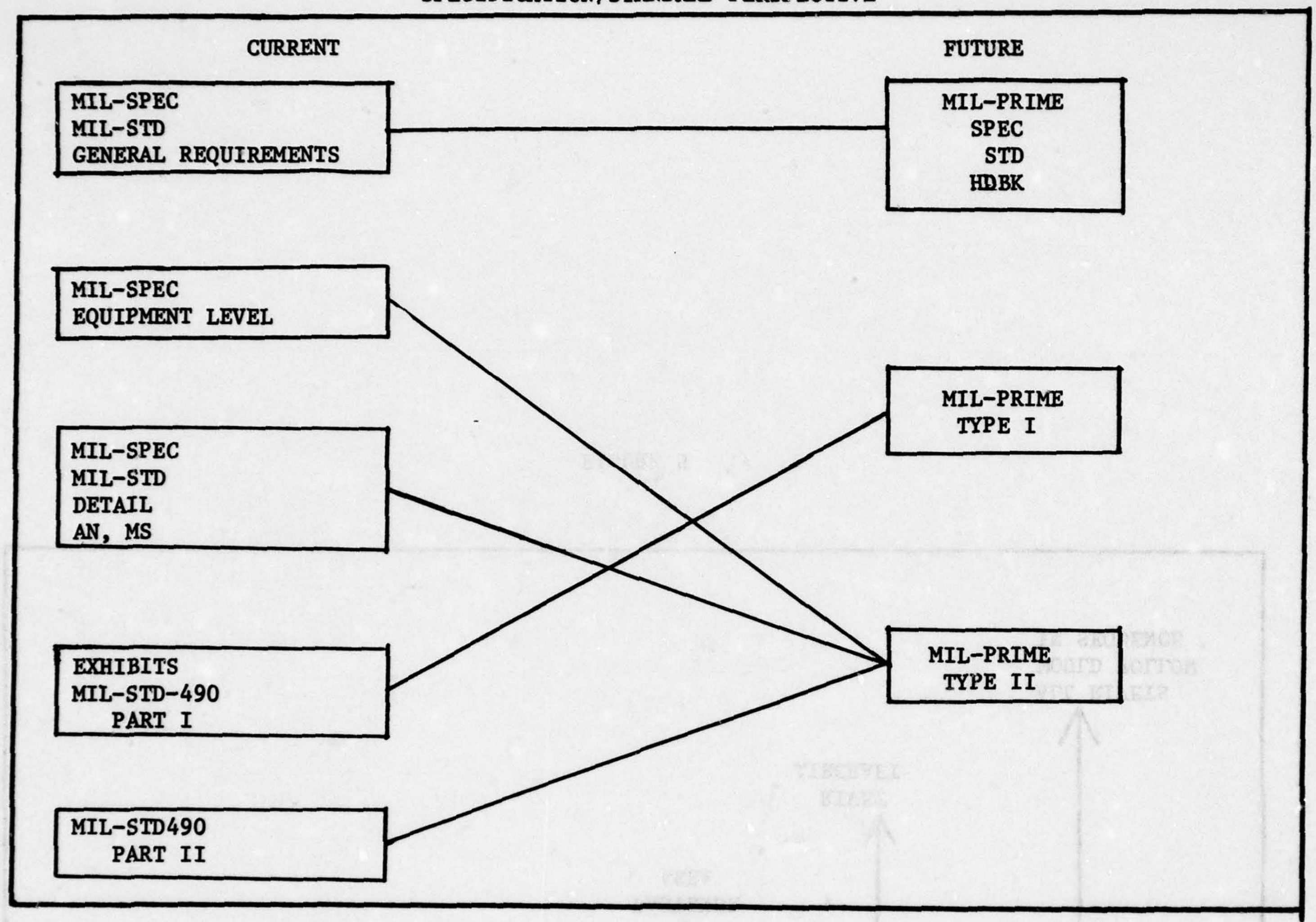

FIGURE 4

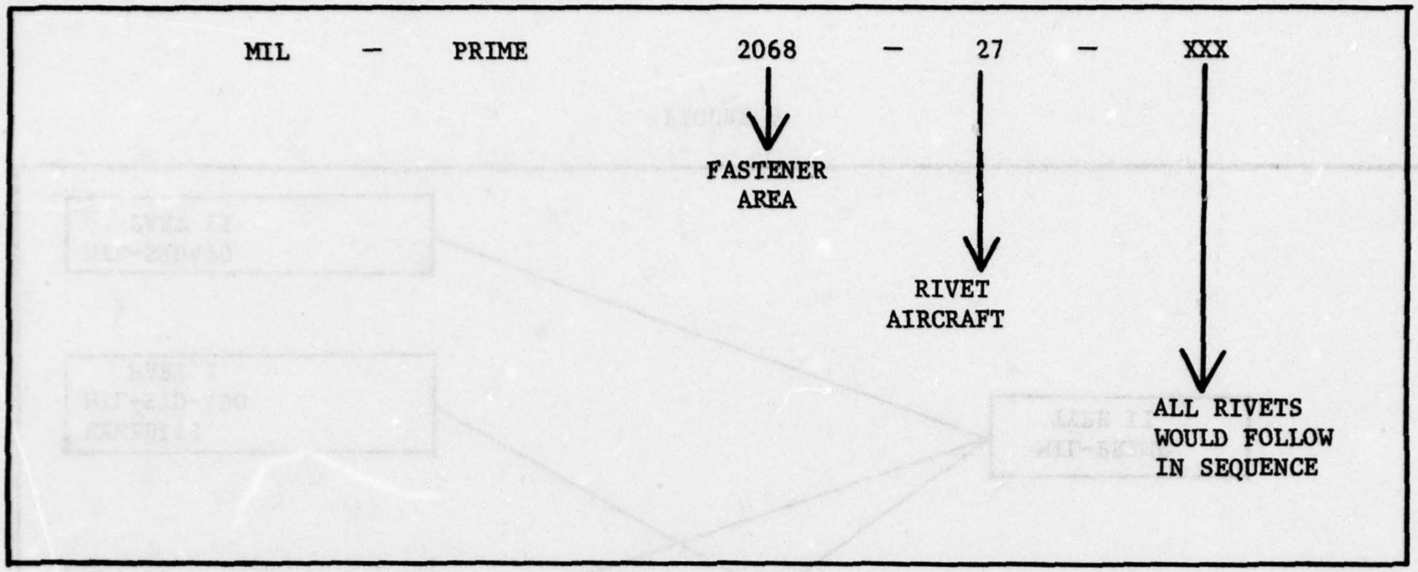
MIL – PRIME 2068 – 27 – XXX
FASTENER AREA
RIVET AIRCRAFT
ALL RIVETS WOULD FOLLOW IN SEQUENCE

FIGURE 5

SUGGESTIONS FOR THE MIL-PRIME-STANDARD ORGANIZATION

(Informal Report)

E. Frank Carlson
The Boeing Company

The forthcoming MIL-PRIME-STD and MIL-PRIME-HANDBOOK offer potential for significant improvement in the specification of flying qualities requirements. The MIL-PRIME-STD will give the basic framework for the specification of flying qualities. Suggestions and rationale for detailed requirements will be given in the MIL-PRIME-HANDBOOK.

Each SPO will be able to tailor the flying qualities requirements to the specific needs and objectives of its programs. The new MIL-PRIME-STD/-HANDBOOK should also facilitate keeping the specifications abreast of a rapidly changing Technology. In my opinion, many of the flying qualities specification problems encountered during the AMST (Advanced Medium STOL Transport) program would have been alleviated by the proposed MIL-PRIME-STD/-HANDBOOK concept.

In creating this new MIL-PRIME-STD it would be very desirable, from the airplane designer's viewpoint, to rearrange the flying qualities topics relative to the order in which they now appear in MIL-F-8785B. The organization used in MIL-F-8785B follows primarily along the lines of flight phases and the individual parameters used to specify flying qualities. This organization tends to be very difficult for the flight control system designer to use. For example, when designing the pitch axis feel system the designer must compile a list of all requirements related to the pitch axis control forces and displacements. In reviewing MIL-F-8785B he will find that these requirements are scattered through sixteen separate sections of the specification, ranging all the way from section 3.2.2.2.1 to section 3.6.3.1. This type of problem prompted Boeing to evaluate alternate organizational structures when the Design Requirements and Objectives document was written for the YC-14.

The table of contents showing the organization of the YC-14 Design Requirements and Objectives document is given on the following pages. The subjects have been grouped by airplane systems to simplify the control system design tasks. A similar arrangement is recommended for the MIL-PRIME-STD.

3.1 GENERAL REQUIREMENTS AND DEFINITIONS

3.1.1 Operational Philosophy

- 3.1.1.1 Operational Modes
- 3.1.1.2 Flight Phase Categories
- 3.1.1.3 Operating Margins
- 3.1.1.4 State of the Aircraft

3.1.2 Configurations and Loadings

3.1.3 Flight Envelopes

- 3.1.3.1 Operational Flight Envelopes
- 3.1.3.2 Service Flight Envelopes
- 3.1.3.3 Permissible Flight Envelopes

3.1.4 Flying Qualities Levels and Application

- 3.1.4.1 Aircraft Normal State Flying Qualities
- 3.1.4.2 Aircraft Failure State Flying Qualities
  - 3.1.4.2.1 General Failure State Requirements
  - 3.1.4.2.2 Specific Failure States
  - 3.1.4.2.3 Special Failure States

3.1.5 Atmospheric Environment

- 3.1.5.1 Mean Wind
- 3.1.5.2 Turbulence
- 3.1.5.3 Discrete Gust Analysis

3.1.6 Flight Control Systems

- 3.1.6.1 Primary Flight Controls
- 3.1.6.2 Secondary Flight Controls
- 3.1.6.3 Basic/Manual Modes and Subsystems
- 3.1.6.4 Pilot-Assist/Automatic Modes and Subsystems

3.1.7 System Status Indication

- 3.1.7.1 Advisory
- 3.1.7.2 Caution Indication
- 3.1.7.3 Warning System

3.2 LONGITUDINAL FLYING QUALITIES
3.2.1 Control Power

3.2.1.1 In-Flight Maneuvers
3.2.1.2 Takeoff
3.2.1.3 Landing

3.2.1.3.1 Static Balance
3.2.1.3.2 Pitch Acceleration
3.2.1.3.3 Maneuver Control Power
3.2.1.3.4 Glideslope Control

3.2.1.4 Stall-Recovery
3.2.1.5 Trim Authority

3.2.2 Control Forces and Displacements

3.2.2.1 Column Travel
3.2.2.2 Breakout Forces
3.2.2.3 Constant Speed Maneuver Force and Displacement Gradients

3.2.2.3.1 Stick Force Per g
3.2.2.3.2 Displacement Per g
3.2.2.3.3 Linearity

3.2.2.4 Minimum Stick Forces
3.2.2.5 Maximum Stick Forces

3.2.2.5.1 Trim Changes
3.2.2.5.2 In-Flight Maneuvers
3.2.2.5.3 Accelerated Flight
3.2.2.5.4 Takeoff
3.2.2.5.5 Landing
3.2.2.5.6 Sideslips

3.2.3 Stability and Response Characteristics

3.2.3.1 Minimum Damping
3.2.3.2 Pitch Rate Characteristics
3.2.3.3 Pitch Attitude Hold
3.2.3.4 Flight Path Characteristics

3.2.3.4.1 Hands-Off Flight Path Stability
3.2.3.4.2 Hands-On Flight Path Stability
3.2.3.4.3 Pitch Attitude - Flight Path Relationship
3.2.3.4.4 Load Factor Response

3.4 FLIGHT CONTROL SYSTEMS DESIGN
3.4.1 General Design

3.4.1.1 Flight Deck Controls
3.4.1.2 Powered Control Systems
3.4.1.3 Fail Safety
3.4.1.4 Structural Load Path
3.4.1.5 Dissimilar Load Path
3.4.1.6 Residual Oscillations
3.4.1.7 Stability Margins
3.4.1.8 Endurance Life
3.4.1.9 Warmup
3.4.1.10 Failure Transients
3.4.1.11 Lightening Strikes and Static Atmospheric Electricity

3.4.2 Primary Flight Control System

3.4.2.1 Mechanical Flight Control Elements

3.4.2.1.1 Design for Murphy's Law
3.4.2.1.2 System Load Criteria
3.4.2.1.3 Control Surface Physical Damage
3.4.2.1.4 Control Harmony
3.4.2.1.5 Control Centering
3.4.2.1.6 Power System Checkout
3.4.2.1.7 Segregation of Systems
3.4.2.1.8 Actuator Sizing
3.4.2.1.9 Synchronization of Multiple Actuators
3.4.2.1.10 Control Surface Stops
3.4.2.1.11 Bias and Centering Springs
3.4.2.1.12 Control Surface Actuation

3.4.2.2 Control Augmentation System Elements

3.4.2.2.1 CAS Control Authority
3.4.2.2.2 Gain Scheduling
3.4.2.2.3 Automatic Pitch Trim
3.4.2.2.4 CAS Failure Logic
3.4.2.2.5 Transient Power Effects
3.4.2.2.6 Computers

3.4.2.3 Propulsion Control

3.4.3 Secondary Flight Control System

3.4.3.1 General Criteria

3.4.3.1.1 Flight Deck Controls
3.4.3.1.2 Asymmetric Conditions
3.4.3.1.3 Emergency Operation
3.4.3.1.4 Stabilizer Actuation
3.4.3.1.5 Flap Actuation
3.4.3.1.6 Flap Retraction

3.3 LATERAL-DIRECTIONAL FLYING QUALITIES

3.3.1 Control Power

- 3.3.1.1 Roll Maneuvers
- 3.3.1.2 Yaw Control
- 3.3.1.3 Crosswind Takeoff and Landing
- 3.3.1.4 Asymmetric Thrust
- 3.3.1.5 Trim

3.3.2 Control Forces and Displacement

- 3.3.2.1 Control Travel
- 3.3.2.2 Breakout Forces
- 3.3.2.3 Control Forces
- 3.3.2.4 Turn Coordination
- 3.3.2.5 Control Sensitivity
- 3.3.2.6 Control Linearity

3.3.3 Stability and Response Characteristics

- 3.3.3.1 Roll Mode Response
  - 3.3.3.1.1 Roll Attitude Command/Hold
  - 3.3.3.1.2 Roll Rate Command
  - 3.3.3.1.3 Bank Angle Oscillations
  - 3.3.3.1.4 Roll Rate Oscillations
- 3.3.3.2 Heading Control
- 3.3.3.3 Turn Coordination
- 3.3.3.4 Roll-Sideslip Stability
- 3.3.3.5 Roll-Sideslip Coupling
- 3.3.3.6 Pilot Induced Oscillations

3.3.4 Stalls

3.3.5 Pilot Assist Modes

- 3.3.5.1 Heading Hold
- 3.3.5.2 Heading Select
- 3.3.5.3 Automatic VOR/TACAN Navigation
- 3.3.5.4 Category II Landing Approach
- 3.3.5.5 Flight Director Modes

3.4.3.2 Trim Systems

3.4.3.2.1 Trim Cockpit Controls
3.4.3.2.2 Trim Using Primary Control Surfaces
3.4.3.2.3 Series Trim
3.4.3.2.4 Effects of Trim System Failures on Primary Controls

3.4.4 Standby Flight Control System

3.4.4.1 Design Simplicity
3.4.4.2 Engagement
3.4.4.3 Override Capability
3.4.4.4 Isolation

3.4.5 Maintainability

3.4.5.1 Operational Checkout Provisions
3.4.5.2 Malfunction Detection and Fault Isolation Provisions

3.4.5.2.1 Built-In Test Equipment
3.4.5.2.2 Provisions for Checkout with Portable Test Equipment

3.4.5.3 Maintenance Personnel Safety Provisions

Wayne Thor, ASD: Do you currently use MIL-F-8785B as a design Handbook?

Answer: No, we do not use MIL-F-8785B as a design handbook. We design a flight control system which we believe will meet the requirements of the spec. Then we evaluate this flight control system against the requirements. We will have to go through the same process with the new MIL-PRIME-STD. Nothing will have changed in this regard.

We have successfully used the proposed organization for the YC-14 Design Requirements and Objectives (DR&O) document. I have shown a summary of the way the DR&O was organized - a complete table of contents is shown in my paper. We found that all of the requirements contained in MIL-F-8785B can be melded into the proposed organization scheme. The first attempt at a DR&O for the YC-14 was organized much like MIL-F-8785B. This caused significant problems for the designers to use.

Hansel Stegall, NASA JSC: Aren't you going to have the same problem with your proposed organization as you accuse the spec? Why rewrite the spec to conform to a new organization? Why not add a cross-reference index at the back?

Answer: No, I do not think it would be satisfactory to simply provide a cross index of requirements to relate subsystem design (such as a feel system) to the various sections of the MIL-PRIME-STD. This would be clumsy. It is not obvious why it is necessary to stick with the same organization as currently in MIL-F-8785B. Some definite improvements are possible.

This request to reorganize the flying qualities requirements is perhaps more significant to Boeing than to some other aerospace companies. Some of our engineers have not spent their entire career working with MIL-F-8785B. It is a definite problem for them to use this spec, since they must spend so much time reading through the various sections to find the ones that apply to their particular design problem(s). This is well illustrated by the pitch feel system where the designer must compile the requirements from 16 major sections of

MIL-F-8785B where design values are given. It should not be necessary to do this sifting and sorting. It leads to an increased probability of oversight and errors in addition to a waste of good engineering manpower. The designer should be able to go to the table of contents in the MIL-PRIME-STD and quickly identify all applicable requirements. This is not possible with the MIL-F-8785B organization.

A NEW APPROACH TO QUANTIFICATION OF FLYING QUALITIES EXCELLENCE

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SUMMARY

Flying qualities specifications have been increasing in complexity and in volume of content over the years. However they specify most requirements in terms of open loop criteria when what is desired is achievement of acceptable closed loop flight performance. This results in increased aircraft procurement costs both because of extensive flight testing necessary, and because some very cost effective means of improving closed loop flight characteristics which do not affect open loop characteristics are precluded from use by current ground rules. An example means of putting a lead lag function in series with pilot input is shown to improve the longitudinal handling qualitites of Lockheed's flight research powered sailplane by about three points on the Cooper-Harper scale. It has no effect on the open loop flight characteristics.

Data are presented illustrating a proposed approach to specifying flying qualities criteria in terms of required pilot reserve attention during the accomplishment of specified portions of each aircraft's mission such as landing approach. Required pilot performance at each level of reserve attention is specified in terms of an easily measured parameter such as his RMS altitude variations from the ideal glide path and correlated against critical performance items such as aircraft rate of sink at touchdown. It is shown that it now appears to be possible to specify rational flying qualities criteria in terms of parameters directly related to desired mission performance without constraining the means by which they are met.

INTRODUCTION

Signal Corps Specification No. 486 dated 23 December 1907 for the first heavier-than-air flying machine purchased by the U. S. Government included direct and to the point flying qualities requirements shown in Figure 1. They may be summarized by: "It should be sufficiently simple in its ---- operations to permit an intelligent man to become proficient in its use within a reasonable length of time." A single mission flight demonstration task was specified. In later years after the issuance of this specification, which was written in terms of the true end product desired, attention began to be devoted to defining quantifiable aircraft characteristics which, when attained, would ensure the intent of the 1907 specification. Requirements were developed for airplane control power to allow it to maneuver as desired and for its stability, primarily to minimize the pilot's workload of making continual flight path corrections in response to disturbances, imprecise control, etc. This has been a difficult and continuing task, largely because

of the very complex interactions between the many parameters which combine to make up what is called an airplane's "flying qualities" - a loose term in itself. The result has been that over the years these requirements have snowballed into a large volume of generally sufficient but at times, unnecessarily expensive conditions for the attainment of good flying qualities.

Immediately after World War II aircraft were being designed to USAF specification R-1815A which was soon followed by R-1815B. In 1954, the first issue of MIL-F-8785 emerged followed by several amendments, and it was superceded in 1968 by MIL-F-8785A. Only a year later, it too was superceded by MIL-F-8785B and its subsequent amendments. One of the first things that impresses one who compares those documents is that as time goes on, they get thicker. R-1815B has 24 pages; MIL-F-8785B has 88 pages not counting the appendix volume of backup or clarification data. Further examination shows that many specific requirements in MIL-F-8785B demand much more complex analyses and flight testing to demonstrate compliance than do their counterparts in R-1815B or in the initial issue of MIL-F-8785. It is estimated that the cost of analysis and simulation effort against MIL-F-8755B compared to MIL-F-8785 has increased by 40 to 50 percent and that the flight hours required for data acquisition have increased by 25 to 30 percent.

However, significant as is this added expense in aircraft development, it is not the most serious difficulty inherent in MIL-F-8785B. For reasons noted earlier, most of the requirements in the specification are "open loop", i.e., they do not specify what the airplane should be able to do with the pilot guiding it, they specify airplane characteristics without pilot participation that are supposed to cause it to meet his handling qualities desires. In Lockheed's opinion, this feature unnecessarily constrains the designer in his selection of the lowest cost means of providing an air vehicle to the customer which embodies satisfactory handling qualities through use of systems of suitable reliability and maintainability. This point is illustrated by a discussion of a recently developed concept for significant improvement of the flyability of an aircraft without modification of its conventionally defined control, stability, or dynamic characteristics.

CONTROL COMPENSATION

It is believed that future generations of energy efficient aircraft will incorporate relaxed static stability principles to an ever increasing extent, in addition to other CCV concepts. The conventional approach to development of desirable flying qualities in such aircraft is the addition of stability augmentation systems which monitor and modify the aircraft's response to control or external inputs and provide control inputs in <u>parallel</u> with those of the pilot to cause the net aircraft response to be satisfactory to the pilot. An alternative approach to providing improved flying qualities is to add a compensating function in <u>series</u> with the pilot inputs to create the same net airplane response as did the stability augmentation system. This function is carefully chosen to compensate for the difference between the airplane's natural response to a pilot's input and that which research has shown is most pleasing to a pilot. For example, a lead function is used where the aircraft

has a large moment of inertia and tends to have a sluggish response. Correspondingly, a lag function may be used where the aircraft's normal response may be too rapid for ease of normal flying such as the case of an RPV or any light, small aircraft. This tailoring of the pilot's input to the aircraft's dynamic characteristics is called "Control Compensation". Figure 2 shows a simple block diagram of a hovering VTOL comparing the mechanization and effect of these two approaches to flying qualities enhancement systems. On the left is shown a simple feedback stability augmentation system which combined with the neutrally stable basic airplane transfer function of the form shown yields the transfer function, θ/δ_C . On the right is shown the equivalent control compensation system which when combined with identical basic airplane characteristics yields the identical net transfer function as does the stability augmentation system. Examination of the control compensation term shows it to be a simple lead control function. Exact duplication of stability augmentation results is not possible for the more complex flight cases where aerodynamic stability and damping are present but a close match is generally possible.

Figure 3 shows the results of a simple three degree of freedom flight simulation exercise. The pilot was presented a displayed attitude command θ_c on a scope. The command varied for each case as shown on the top of the diagram over a period of about one minute. A typical medium transport was being simulated and its damping ratio was varied as shown. Airplane attitude was also displayed to the pilot and his task was to keep the actual airplane attitude consistent with the commanded value. The figure shows that without control compensation, the pilot did very well at the higher values of damping. In the lightly damped case where $\zeta = 0.1$, pilot performance was fair to poor; in the negatively damped case, real control was not achieved. The effect of control compensation is seen to be slight on the cases with adequate damping. In the lightly damped case, the control excursions are markedly diminished and some reasonable measure of control is retained even in the presence of negative damping.

The Control Compensation principle has been evaluated in flight simulations of a number of other Lockheed aircraft. It was also recently flight tested in Lockheed's flight research Caproni jet powered sailplane shown in Figure 4. A sidestick controller was installed and the flying qualities evaluated first with the basic system against those of the unmodified Caproni; then with varying characteristics and extents of control compensation. Figure 5 shows that the flying qualities of the basic unmodified airplane at the aft C.G. limit were rated to be quite poor. This was true both with and without the sidestick controller installation. Note that here the control compensation function is a lead-lag and that its use produced a pilot rating improvement of about three points on the Cooper/Harper scale.

Studies have shown that Control Compensation can be provided to an aircraft at more than an order of magnitude less cost than can stability augmentation and that the resulting system is several orders of magnitude more reliable per channel than is a stability augmentation system. It can be mechanized using either electrical or hydromechanical components hence, has the potential to provide dissimilar redundancy.

While its use may serve as a complete replacement for conventional stability augmentation systems in certain applications, it is not suggested that this is the general case. However, it is believed that the application of this principle in conjunction with other more conventional principles will yield a much less expensive and much more reliable system at no sacrifice at all in system effectiveness. It is again emphasized that Control Compensation affects only pilot-in-the-loop characteristics. Therefore, when flying qualities requirements are specified in terms of airplane characteristics without the pilot-in-the-loop, as do so many of the MIL-F-8785B requirements, the use of this principle is precluded. The Government is thus currently denied the above noted cost and reliability benefits solely because of the manner in which desired flying qualities characteristics are specified.

CLOSED LOOP FLIGHT PERFORMANCE MEASUREMENT

A great deal has been learned about the fundamentals of man/machine interactions and human guidance principles during the years while specifications have continued to identify desired airplane characteristics without human guidance. It is therefore suggested that a totally new approach be considered to the business of specifying flying qualities. Lockheed-Georgia has this year initiated an IRAD Program which has as one of its objectives the evaluation of a concept for specifying flying qualities excellence in terms related only to the pilot's ability to accomplish certain specified tasks. This concept involves the determination of a pilot's performance in accomplishing a task such as landing approach where his performance is measured in terms of quantifiable units such as his RMS deviation from the glideslope. His performance is then measured over a range of percentage time which the pilot must devote to peripheral tasks in order to determine his flying performance variations with his percent reserve attention capability. "Reserve attention" is defined as the percentage of the pilot's time he may safely spend on tasks other than actual flight guidance during the accomplishment of a definable portion of his mission such as landing approach. Implementation of this principle is discussed as follows.

A pilot's duty environment may be represented by a diagram as shown in Figure 6. Beginning at the 12:00 position and proceeding counterclockwise, the first segment indicates those tasks associated with basic flight guidance. If the percentage of his time required to accomplish these tasks does not exceed some value which still allows a fairly large percentage reserve attention capacity, and time to accomplish the other tasks noted, the aircraft exhibits Level One flying qualities. While this definition is not an exact paraphrase of that in MIL-F-8785B, it is certainly consistent with it. Now is for some reason the percentage of the pilot's attention capacity required for the accomplishment of his flight guidance tasks grows until it uses up most or all of his reserve attention and perhaps diminishes the attention he can provide to peripheral tasks such as outside visual checks, one may consider the aircraft to be exhibiting Level 2 flying qualities. Similarly, Level 3 flying qualities involves near total pilot effort in pure flight guidance leaving only minimal capability for other essentials such as crew communication.

Pursuing this approach, flight simulation studies were initiated to measure in definable units the effect of pilot workload on glideslope tracking performance. A typical Lockheed advanced transport aircraft was simulated and its flying qualities were progressively degraded by reducing its effective tail area and by moving its C.G. aft. Thus several different airplane configurations were defined which exhibited Cooper ratings covering virtually the complete scale of flying qualities from excellent to catastrophic. As each configuration at its specific Cooper rating was flown on successive approaches, varying amounts of the pilot's attention were directed away from the flight guidance task. These diversions were set up as percentages of a seriesof eight second cycles during each run. For example, at 50% reserve attention, the pilot would "fly" for four seconds, look away for four seconds, return his attention to flight guidance for another four seconds, etc. All runs began at 1,000 feet altitude above the airfield using a straight in approach. Light turbulence was included, i.e., 1/2 the MIL-F-8785B specified value. Measurements were taken of the actual glide path achieved compared to the ideal intended path and an RMS altitude deviation determined for each run. It thus became possible to plot RMS altitude deviation observed as a function of the percentage of time the pilot's attention is demanded elsewhere; i.e., his reserve attention capacity. Such a plot is shown in Figure 7 with curves for each of several flying qualities levels, or Cooper ratings. Each point shown is the average of a series of five runs. The standard deviation value of the data scatter within each five run series was about a quarter to half the altitude deviation values shown. As would be expected, a rather consistent deterioration in glidepath altitude tracking performance is shown with increasing pilot reserve attention (or percentage distraction) and with progressively poor pilot rating. The curves appear to be roughly parallel at the lower values of pilot reserve attention and the airplane is seen to "get away from" the pilot at progressively lower values of pilot reserve attention as pilot Cooper ratings are degraded. During the one part of the process of data acquisition, the pilot became much more fatigued than anyone realized. Only later when the data were extended (and didn't correlate) was the earlier fatigue situation recognized. Some of the "tired pilot" points are shown by flagged symbols on Figure 7; considerable performance degradation is evident.

The real concern related to glidepath deviation, of course, is not with the deviation itself, but in its influence on probable touchdown rate of sink or touchdown position on the runway compared to that intended. For purposes of this analysis, the former was chosen. Figure 8 presents a plot of the maximum touchdown rate of sink achieved in each set of five runs versus the average RMS glidepath deviation. The philosophy is that what is being sought is a measure or indicator of the poorest probable landing for each condition investigated. It is seen that for the P.R. = 2.5 aircraft, no definable trend of touchdown rate of sink with altitude deviation appears to exist, whereas for the poorer pilot ratings, a very clear trend is evident. This may be partially due to the fact that the particular configuration simulated exhibited a strong positive ground effect. Hence the stable configurations tended to a natural flare on landing. The P.R. = 2.5 aircraft showed essentially the same touchdown sink rate <u>hands off</u> as it did in the poorer of the piloted landing cases. Since most transport aircraft are

designed for a limit touchdown rate of sink of 9 feet per second, the minimum level of RMS deviation which yields 9 fps touchdown sink rate has been chosen as the outer value of a Level 3 flying qualities band. Similarly, a desirable rate of sink is 2–3 fps, however, the pilot was unable to achieve this value consistently with even the best of the configurations flown. Hence, the lowest RMS deviation on the scatter-band which yields a 4 fps touchdown rate has been selected as the maximum acceptable for Level 1 flying qualities. The boundary for Level 2 flying qualities has for purposes of this plot been arbitrarily located halfway between the Level 1 and the Level 2 boundaries. This plot has now defined limit values of RMS glideslope deviations for each flying qualities level.

By noting the RMS altitude deviation values included in each flying qualities level band for each pilot rating, the previous two plots may now be combined to yield Figure 9. This plot is in a form suitable for inclusion in a specification. It specifies readily determinable airplane performance against a specific task and the requirements for each flying qualities level are clearly identified. It is seen that the distinction between the flying qualities levels in terms of altitude deviation becomes vanishingly small at the higher levels of pilot inattention (highest reserve attention). While this is perhaps logical, it is also possible that the shape of these boundaries has been inadvertently perturbed by the aforementioned favorable ground effect which should not be the case for general specifications. Further, it is emphasized that this figure is intended to convey criteria related to only one performance parameter desired, i.e., a reliably safe touchdown. Lockheed expects to expand this investigation during the balance of the year by adding further degrees of freedom/constraints to the basic task such as going to the full six degrees of flight freedom and directing the pilot to attempt each touchdown on a specific point on the runway.

Another critically important measure of a pilot's flight performance on the glide path, significant to longitudinal guidance, is airspeed control. This is important both to maintain adequate stall margin for maneuvering and to accommodate reasonable levels of near ground wind variations, as well as to ensure against exceeding a touchdown airspeed consistent with runway length limitations. To develop this means of performance criteria measurement, data are also taken during the above discussed simulation runs of RMS airspeed error during the landing approach and this is plotted versus pilot reserve attention in Figure 10. Once again, very little difference is observed in pilot performance between the P.R. = 2.5 and the P.R. = 4 aircraft. Although very little difference due to pilot fatigue is noted on the P.R. = 4 points, a drastic change is seen in the P.R. = 6 data. The effect on airspeed deviation in going from P.R. = 4 to P.R. = 6 was considerably less than anticipated for the unfatigued pilot which may be partially due to the simplistic 3 degree of simulation used. Believing that the fatigued pilot data may be more representative of the general case, it will be used at this time until more comprehensive data are run later in the year.

Records are also made of the airspeed at touchdown and of the minimum airspeed encountered during landing approach. A plot is then made as seen in Figure 11 of touchdown airspeed versus RMS airspeed error. Only those airspeed points are plotted where the landing was made at above the desired airspeed. From a performance viewpoint, touchdowns at below the desired airspeed have no adverse significance. Somewhat surprisingly, relatively few data points are available because in these simulations, by far the greatest preponderance of cases yielded a touchdown airspeed that was below rather than above the target value. Also plotted are the points indicating the minimum airspeed encountered in any of each of the five runs made, showing the nearest proximity to stall occurring during the approach to the 50 ft. height. The selection of criteria to be used in establishing the limits of airspeed deviations acceptable within each flying qualities level could be a study in itself and should certainly be specified by the customer. For illustration purposes, the Level 3 boundary was selected to be that point on the minimum airspeed line where stall warning at 1.07 V_S was barely avoided while maneuvering the airplane in a 20 degree banked turn--approximately 1.1 V_S. Landing distance varies approximately as the touchdown velocity squared and all flight handbook values are expected to be accurate within 5%. Therefore, the maximum RMS deviation allowable for Level 1 flying qualities has been specified to be at the point where the maximum touchdown airspeed on the scatterband exceeds the desired by 2 1/2%. Since the target airspeed is held constant during the latter portion of the landing approach to the onset of landing flare, for convenience the airspeed actually shown for this criteria is that encountered at the 50 ft. height. Lastly, as before, the Level 2 boundary is shown halfway between Levels 1 and 3. This figure thus establishes the values of RMS airspeed error at the boundaries of the flying qualities levels.

Figure 12 may now be prepared using the boundary curves from Figure 10, the cutoff values of V_{eRMS} from Figure 11, and by fairing an upper boundary curve. This plot, together with that of Figure 9 are presented as illustrative of an approach towards establishment of sufficient flying qualities criteria for any transport aircraft in the landing approach flight regime. The performance criteria chosen to deliniate flying qualities levels have yielded a surprisingly and perhaps unrealisticly large range of level 1 performance. Based on these data, it would appear that the actual capability of a skilled pilot to accomplish critical tasks is not appreciably affected by flying qualities in the upper four levels of pilot rating and only modestly so through a rating of 6. It is suspected that the difficulty of the flying task used for criteria establishment should be increased, since it appears that flight safety is a significant function of flying qualities only in those cases where piloting skill is taxed by environmental circumstances such as severe turbulence, wind shear, engine failure, obstacle clearance, or by pilot fatigue. As noted earlier, these considerations will be addressed later this year in the Lockheed simulations.

Obviously this approach can be extended to defining piloting tasks for other parts of an airplane's mission profile. Scme of these are indicated in Figure 13. It is noted that the principle embodied in specifying flying qualities in the above noted fashion is not new. It is in fact quite consistent with the flight performance specifications provided by the Air Force for automatic flight control systems such as those shown in Figure 13. In each case rational criteria are presented which are directly related to desired mission performance requirements without constraining the means by which they are met.

PERSPECTIVE

It is worth a few moments to consider those factors which contribute to pilot control of flight. Some of these are shown in Figure 14. In years past, only the first two items indicated were of particular significance and the pilot's inherent capabilities and limitations were very imperfectly understood. Hence, flying qualities requirements were specified in terms of airframe parameters based on entirely empirical data concerning pilot desirements. The idea was to provide the pilot with whatever appeared to make him comfortable in the belief that in so doing flight safety and mission effectivity were being maximized. However, in later years, two very significant developments have occurred and are progressing. (1) First, as everyone knows, automatic flight control systems came on the scene and introduced a whole new dimension of design options not only to obtain completely automated flight, but also to provide virtually any level of flying qualities largely independent of basic airframe characteristics. (2) Secondly, a much deeper understanding of the human pilot's functional capabilities is being achieved in AFCS terms. It is thus possible to mathematically model the pilot, as well as the airframe and the flight control system elements, and then to directly solve for the effect of variations in the characteristics of any element in the total system on the effectivity of the total system. That is why it is now possible to greatly improve the flyability of an airframe with otherwise poor flying qualities by tailoring--making compatible--the pilot's control inputs to the inherent airframe characteristics with simple mechanisms rather than multi-channel AFCS systems.

It is incumbent, therefore, on those who write the specifications against which future USAF aircraft will be procured to be sure that the truly desired airplane characteristics, those which determine the aircraft's ability to perform its mission, are accurately and completely specified in terms of the desired features. Recognition must be maintained that all those design features shown in Figure 14 work together and must be accounted for in the determination of what constitutes suitable flight handling performance. Any other approach is certain to have the dual disadvantages of incompletely assuring attainment of desired performance and of precluding use of the most cost effective design approaches to developing needed vehicle systems.

SIGNAL CORPS SPECIFICATION, NO. 486.

WASHINGTON, D. C., *December 23, 1907.*

ADVERTISEMENT AND SPECIFICATION FOR A HEAVIER-THAN-AIR FLYING MACHINE.

GENERAL REQUIREMENTS.

...and trial flights will be at Fort Myer, Virginia.

8. It should be so designed as to ascend in any country which may be encountered in field service. The starting device must be simple and transportable. It should also land in a field without requiring a specially prepared spot and without damaging its structure.

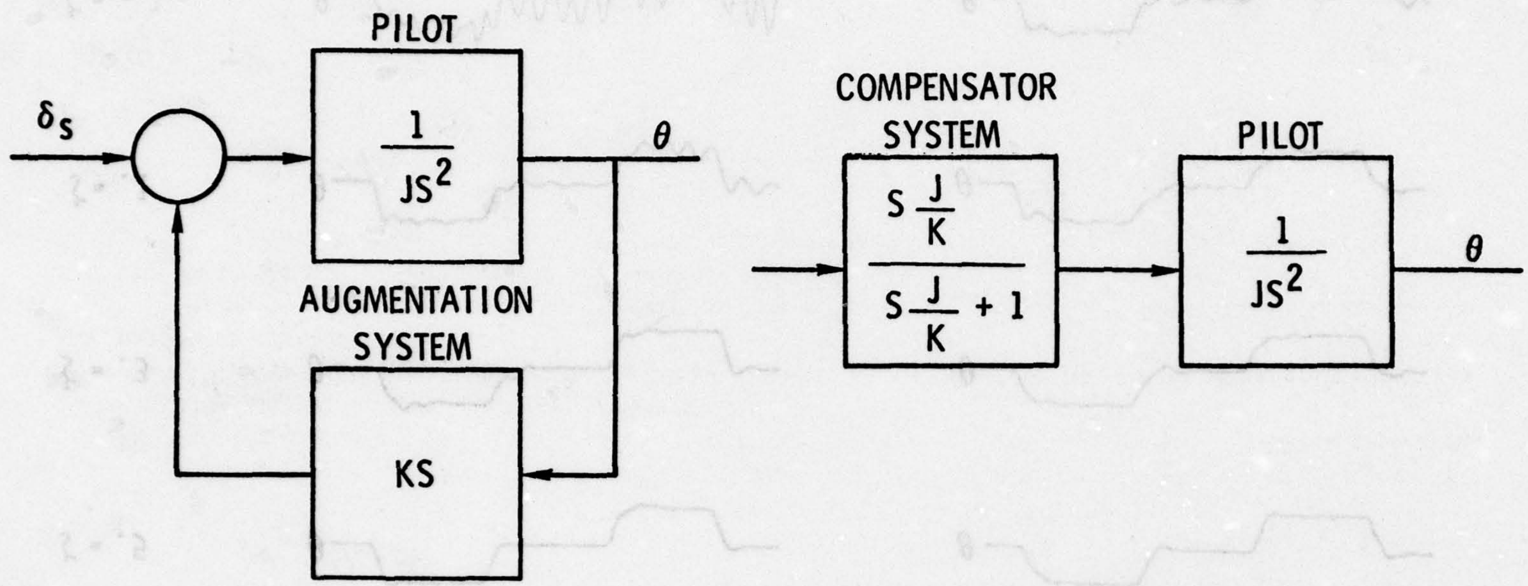
9. It should be provided with some device to permit of a safe descent in case of an accident to the propelling machinery.

10. It should be sufficiently simple in its construction and operation to permit an intelligent man to become proficient in its use within a reasonable length of time.

11. Bidders must furnish evidence that the Government of the [illegible] the lawful [illegible] all patented [illegible]

FIGURE 1

FLYING QUALITIES ENHANCEMENT SYSTEMS



STABILITY
AUGMENTATION

$$\frac{\theta}{\delta_S} = \frac{1}{KS(S\frac{J}{K} + 1)}$$

CONTROL
COMPENSATION

$$\frac{\theta}{\delta_S} = \frac{1}{KS(S\frac{J}{K} + 1)}$$

FIGURE 2

LONGITUDINAL TRACKING RESPONSE

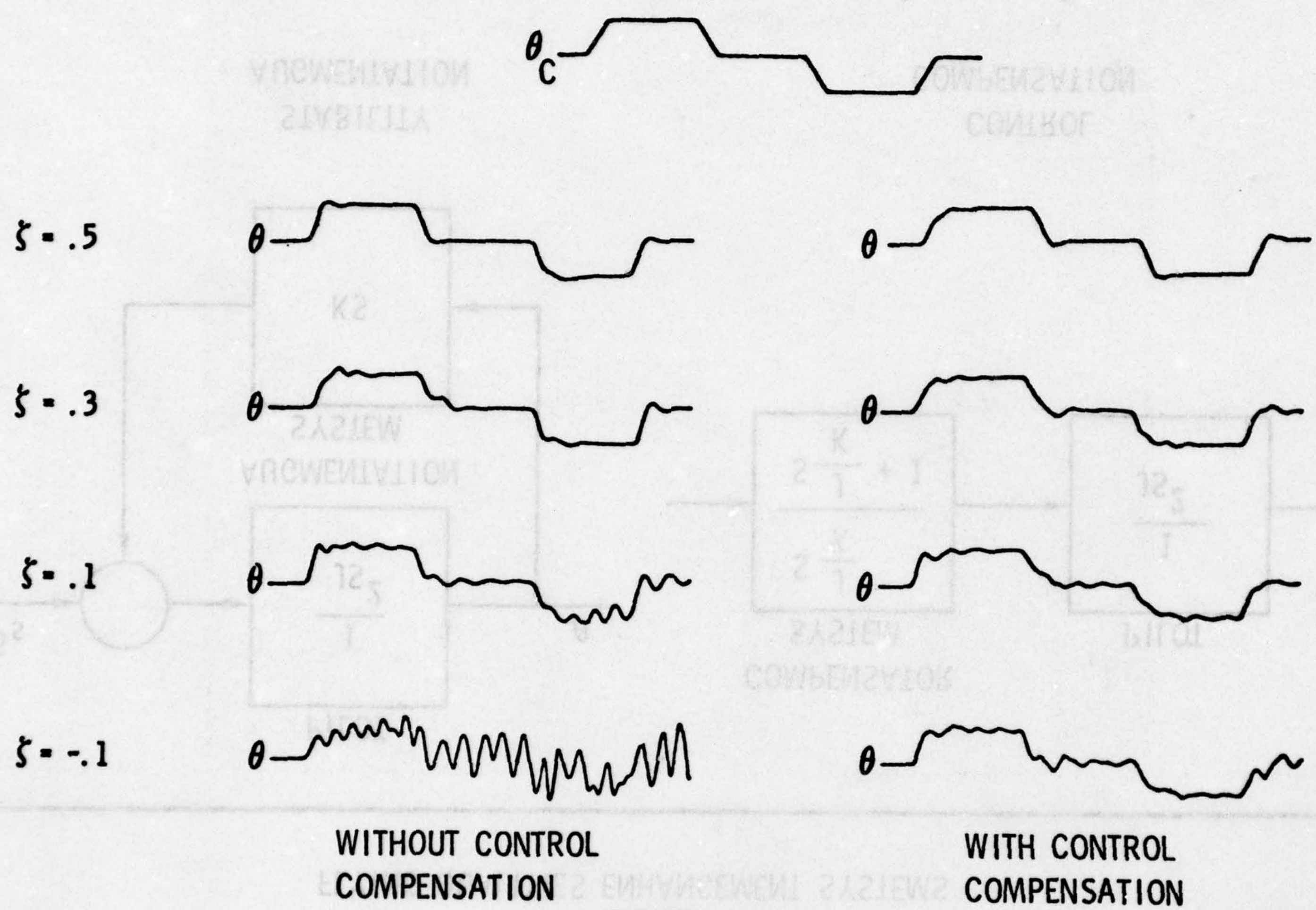


FIGURE 3

LOCKHEED FLIGHT RESEARCH AIRCRAFT

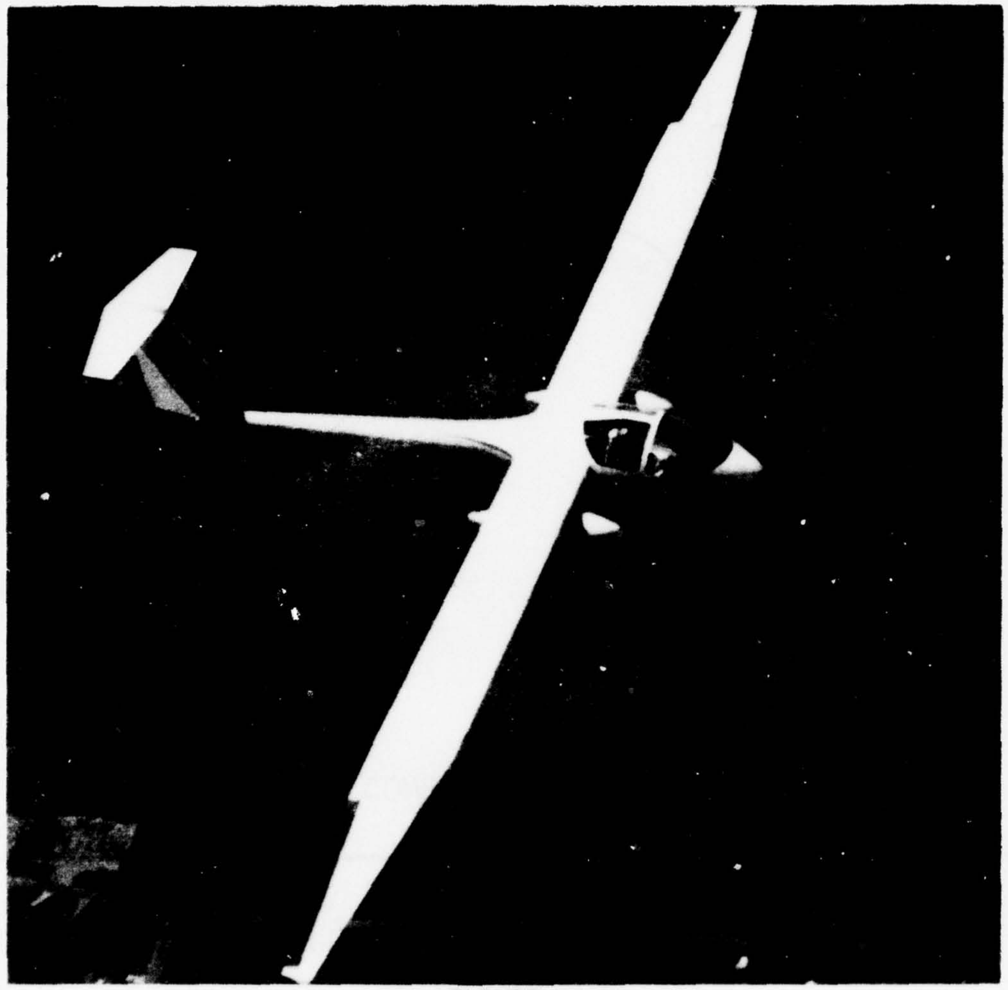


FIGURE 4

EFFECT OF CONTROL COMPENSATION ON CAPRONI FLYING QUALITIES

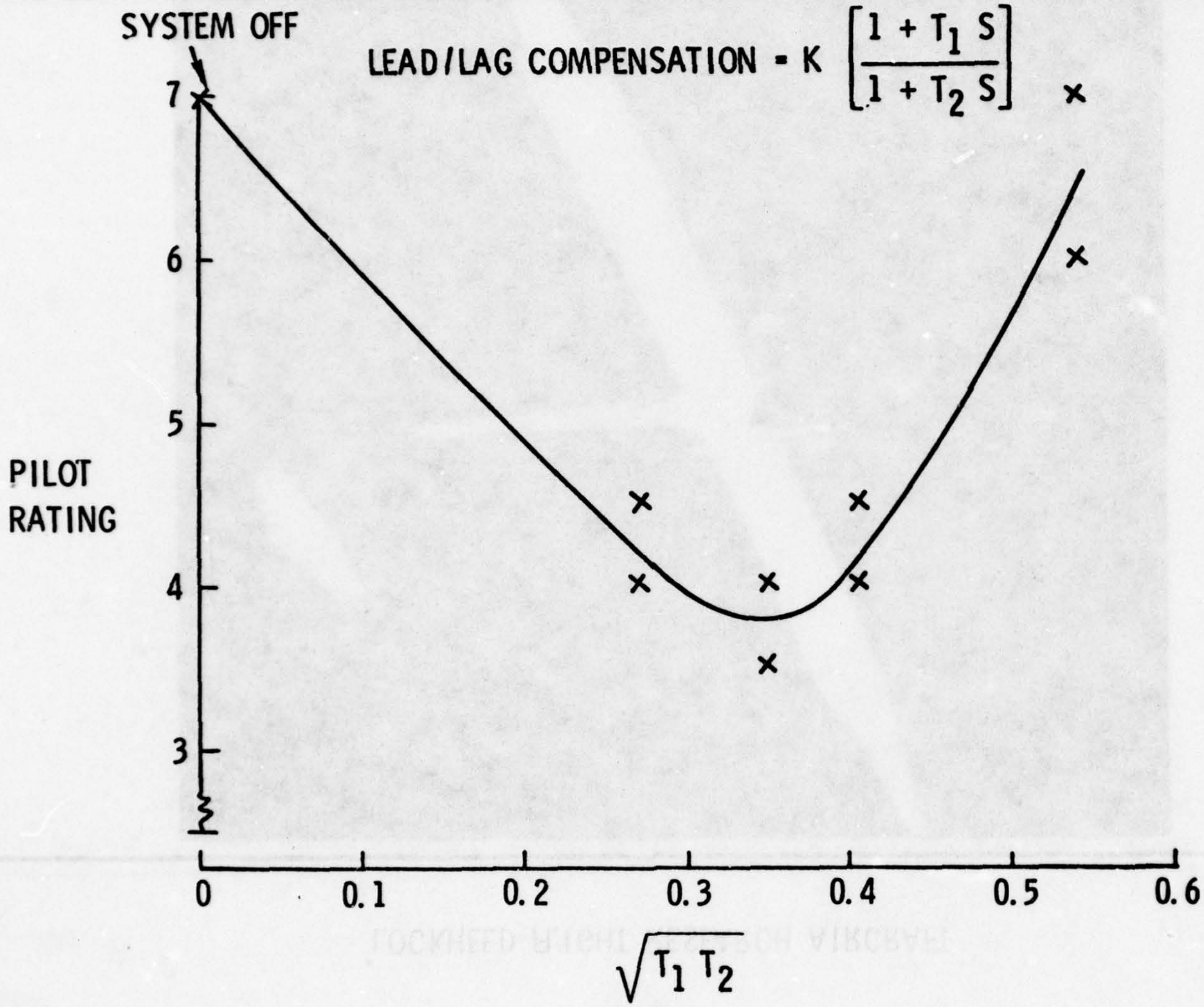


FIGURE 5

TOTAL PILOT WORKLOAD

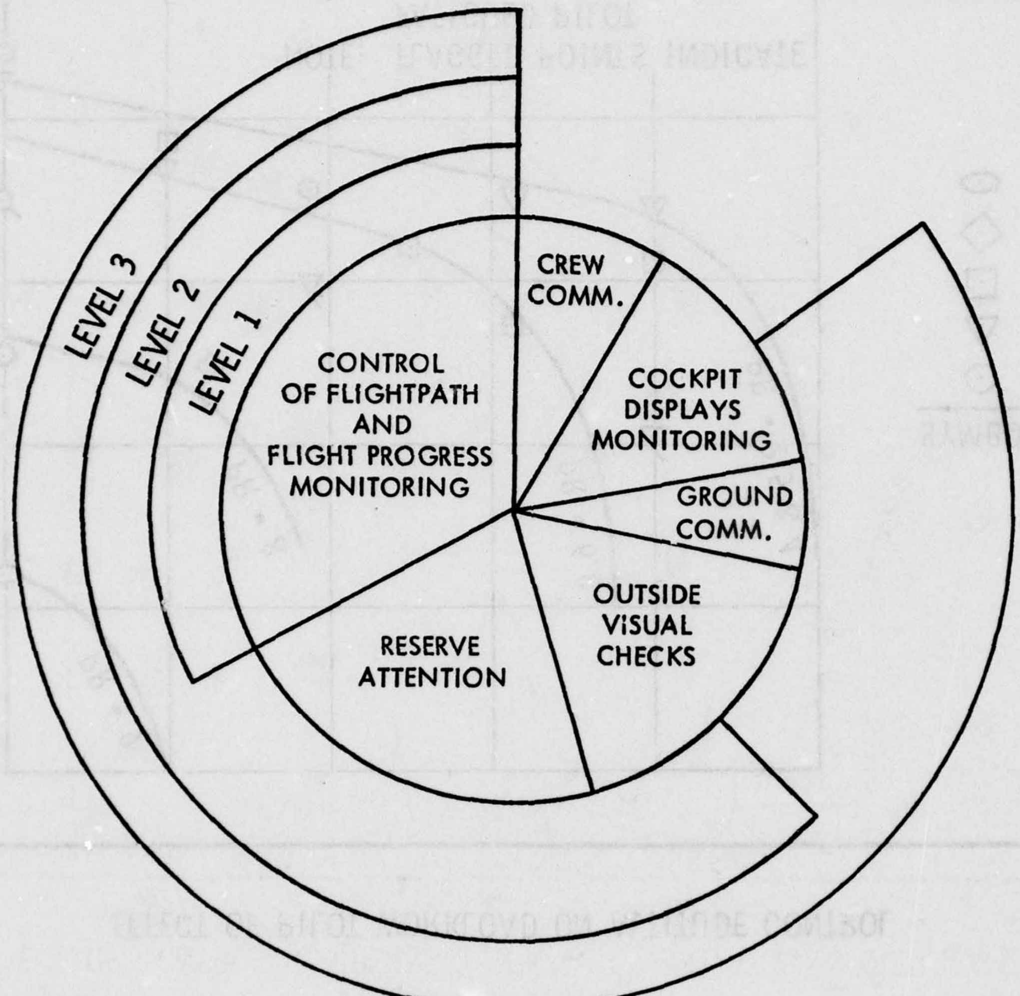


FIGURE 6

EFFECT OF PILOT WORKLOAD ON ALTITUDE CONTROL

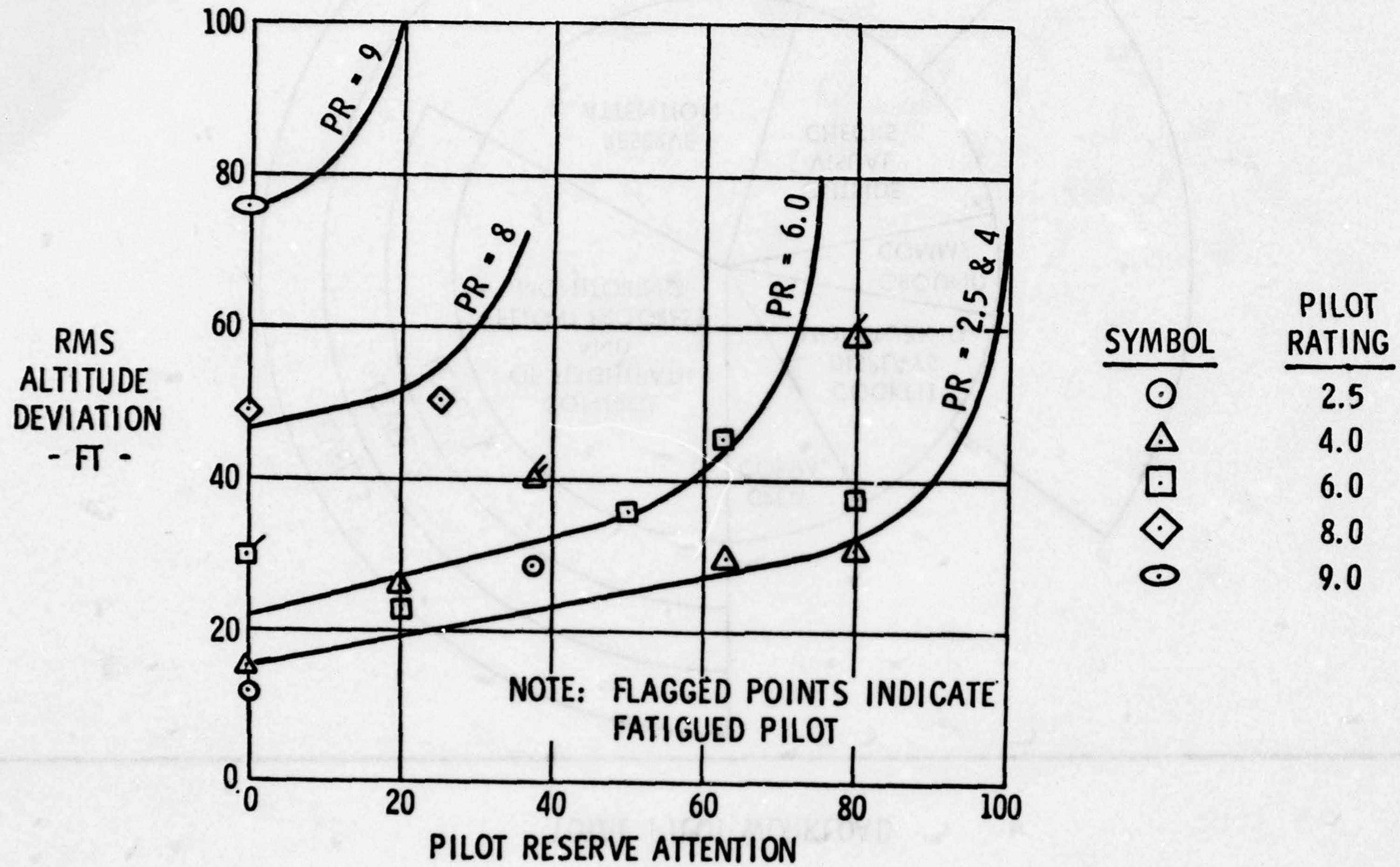


FIGURE 7

CORRELATION OF RMS ALTITUDE DEVIATION WITH LANDING R/S

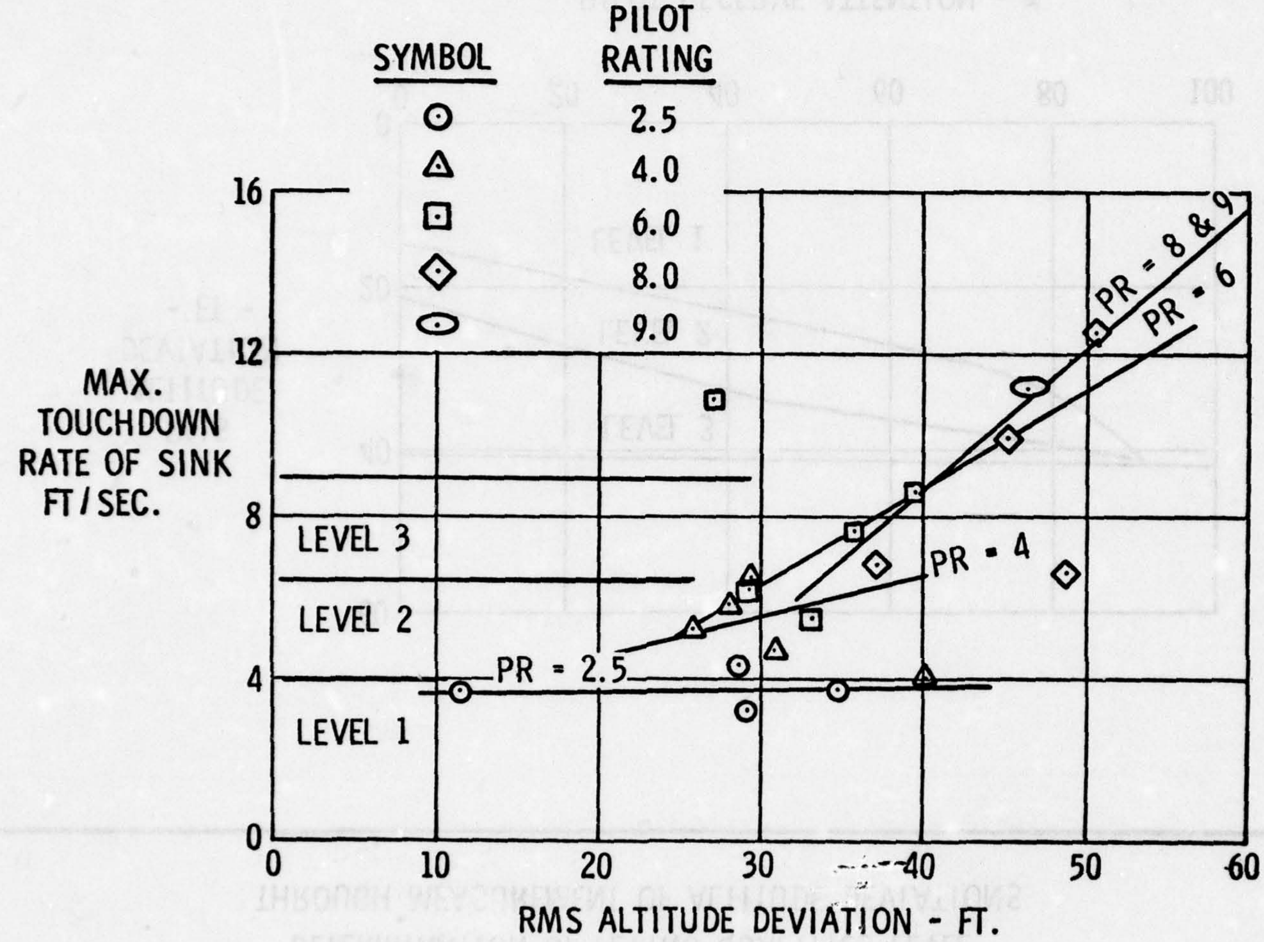


FIGURE 8

DETERMINATION OF FLYING QUALITIES LEVEL THROUGH MEASUREMENT OF ALTITUDE DEVIATIONS

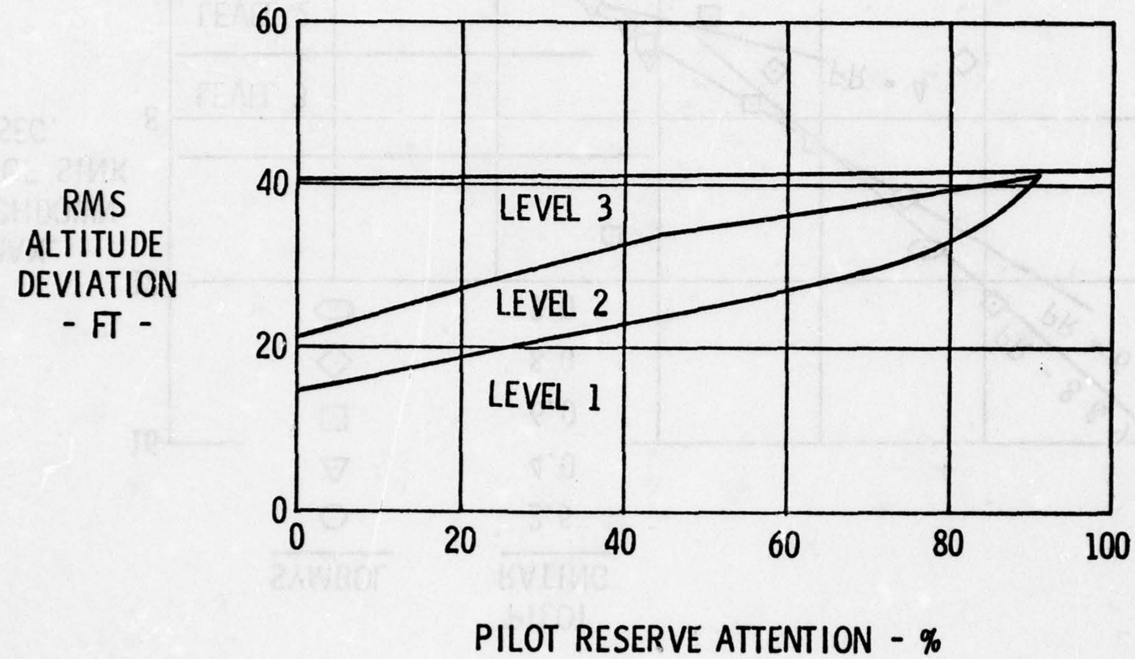


FIGURE 9

EFFECT OF PILOT WORKLOAD ON AIRSPEED CONTROL

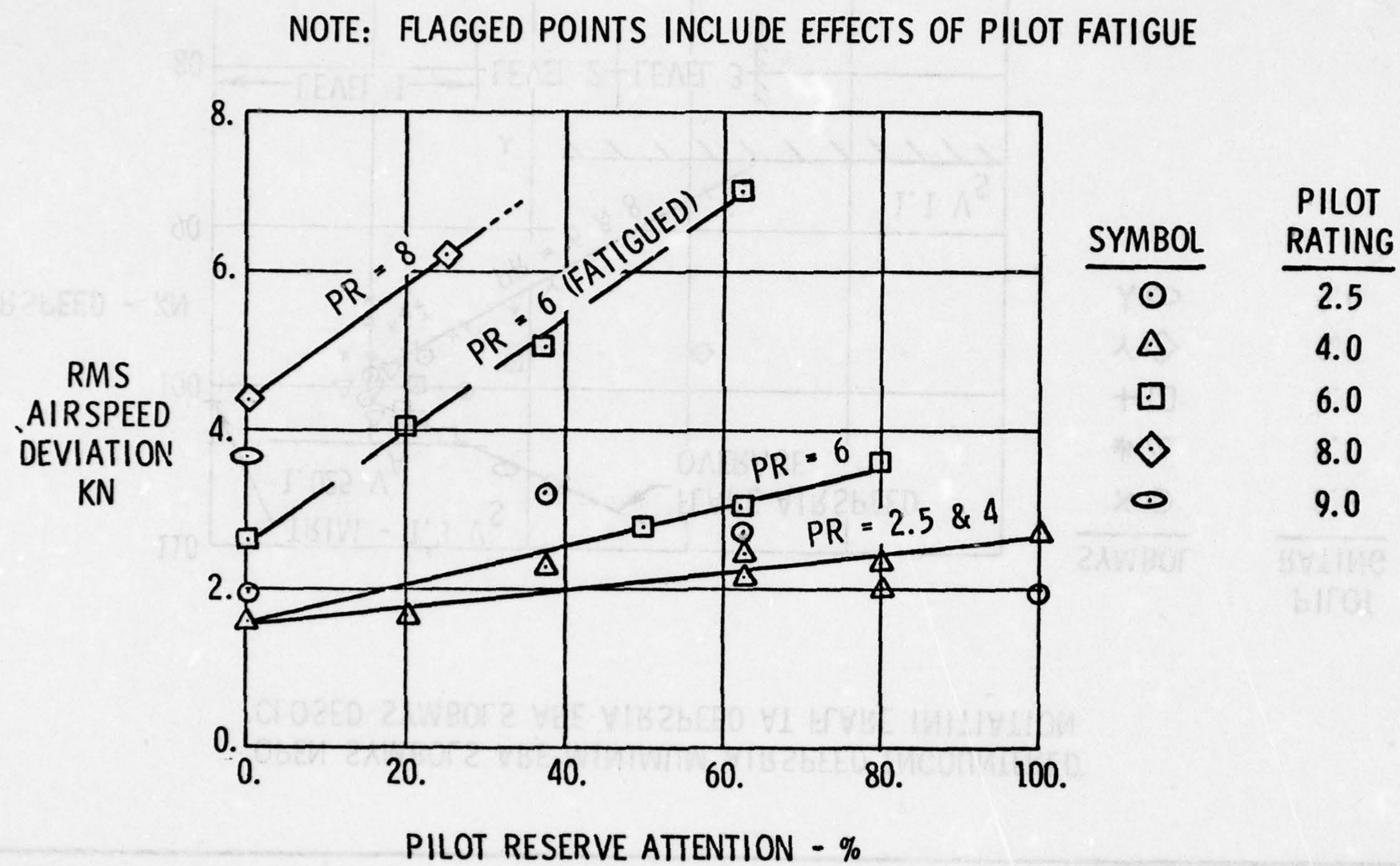


FIGURE 10

GLIDESLOPE AIRSPEED CONTROL

OPEN SYMBOLS ARE MINIMUM AIRSPEED ENCOUNTERED
CLOSED SYMBOLS ARE AIRSPEED AT FLARE INITIATION

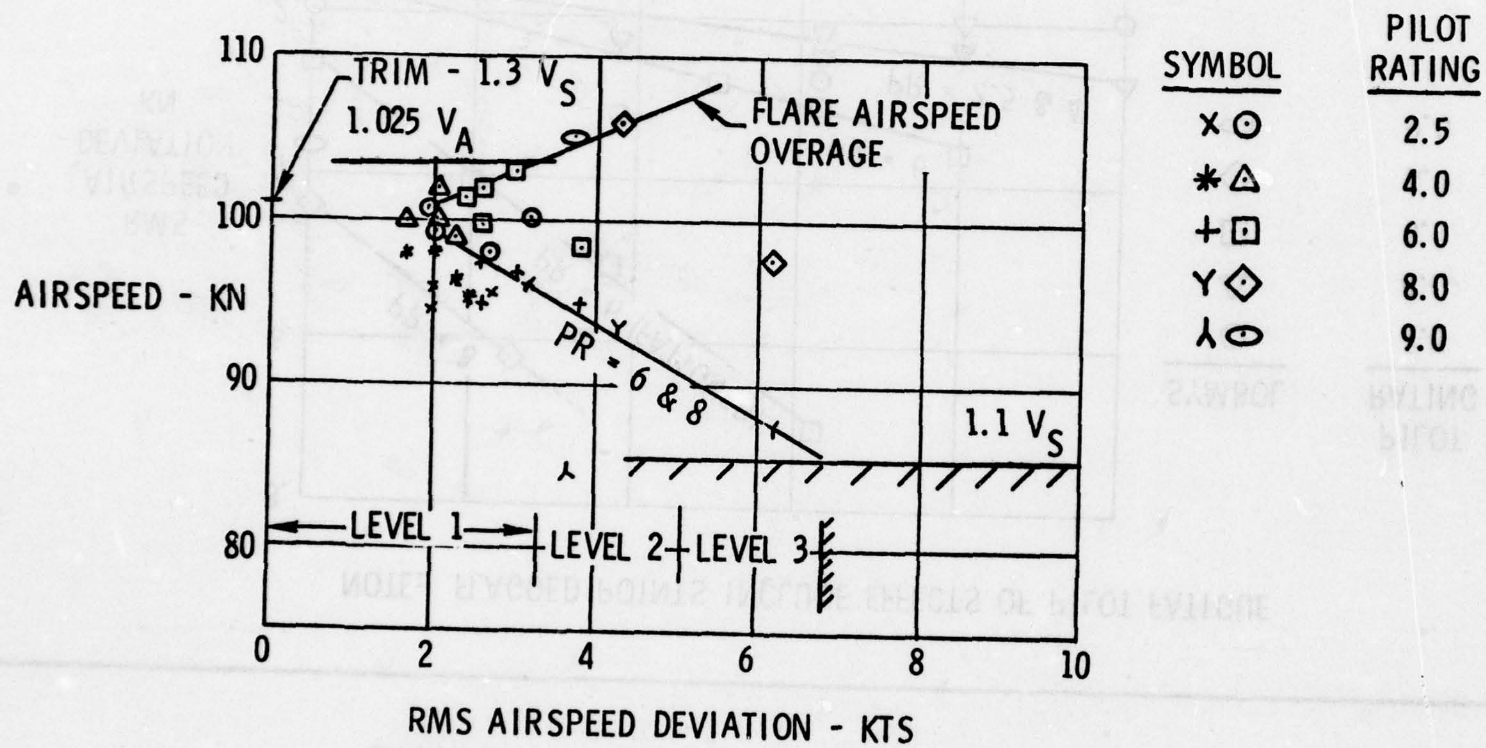


FIGURE 11

DETERMINATION OF FLYING QUALITIES LEVEL
THROUGH MEASUREMENT OF AIRSPEED DEVIATIONS

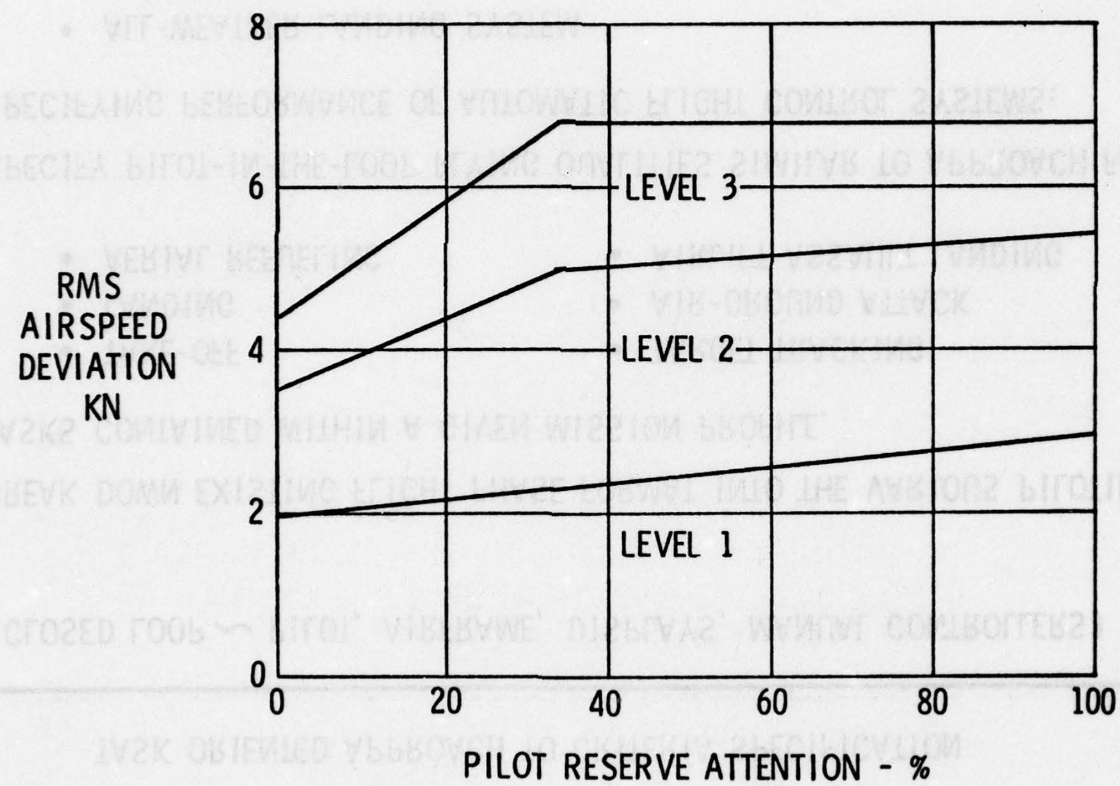


FIGURE 12

TASK ORIENTED APPROACH TO CRITERIA SPECIFICATION

(CLOSED LOOP ~ PILOT, AIRFRAME, DISPLAYS, MANUAL CONTROLLERS)

- BREAK DOWN EXISTING FLIGHT PHASE FORMAT INTO THE VARIOUS PILOTING TASKS CONTAINED WITHIN A GIVEN MISSION PROFILE.
  - TAKE-OFF
  - LANDING
  - AERIAL REFUELING
  - TARGET TRACKING
  - AIR-GROUND ATTACK
  - AIRLIFT ASSAULT LANDING
- SPECIFY PILOT-IN-THE-LOOP FLYING QUALITIES SIMILAR TO APPROACH FOR SPECIFYING PERFORMANCE OF AUTOMATIC FLIGHT CONTROL SYSTEMS:
  - ALL-WEATHER LANDING SYSTEM
  - AUTO-PILOT
  - TERRAIN FOLLOWING
  - TASK ORIENTED CONTROL SYSTEMS
  - FUSELAGE AIMING

FIGURE 13

FACTORS CONTRIBUTING TO PILOT CONTROL OF FLIGHT

- PILOT SKILL AND TRAINING
- AIRFRAME
- FLIGHT CONTROL SYSTEM
  - AUTOMATIC
  - PRIMARY
  - SECONDARY
- FLIGHT PATH DISPLAYS

FIGURE 14

Jerry Lockenour, Northrop: How did you control the pilot attention?

Answer: We required the pilot to look away to another area when a red light came on. When a green light then came on, he resumed his flight task.

Capt Martin, AFFTC: How do you propose to quantify pilot reserve attention safely in flight test?

Answer: Experimentally, obviously in a safe environment such as doing a landing approach to a predetermined altitude at 10,000 ft say.

John Hodgkinson, MACAIR: How do you predict pilot attention analytically?

Answer: We don't. That's part of the experimental objective.

Bill Levison, BBN: There are a variety of models to predict attention. Doesn't % attention depend on cycle time?

Answer: Probably. We tried two different cycle times and settled on 8 seconds, the lesser of the ones used. In the real case, of course, cycle times would vary quite widely.

Jerry, Lockenour, Northrop: The pilot compensation would allow an electric backup control system.

Answer: It could. Even better, it can be mechanized entirely hydraulically which improves its reliability per channel by more than two orders of magnitude, and provides dissimilar redundancy.

Representative from 6570 AMRL: Pursuant to the question from Capt Martin AFFTC, we here at AMRL have a number of nonintrusive procedures for assessing percent of available attention and workload by psychological methods. These involve secondary task, time estimation, and psychophysiological methods. We would be happy to discuss these at length any time.

NEW CRITERIA FOR ADVANCED AIRCRAFT DESIGN

Ralph H. Smith

Norman D. Geddes

14 September 1978

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2800 Indian Ripple Road
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Abstract

The general problem is considered of how the tools of pilot-vehicle systems analysis may be used to develop handling quality specifications for systems design. A commentary is given with supporting analysis and data on the Cooper-Harper and the PIOR scales, their role in the design of experiments, and implications to pilot rating variability. A viewpoint is offered about the intent of MIL-F-8785B and the conflicting requirements posed by procurement versus system design. Elements from classic pilot-vehicle systems analysis methodology are distilled against a theoretical framework for the prediction of Cooper-Harper rating to develop a simplified and practical approach to handling qualities prediction that is directly applicable to the development of design specifications. Motion cue effects on handling qualities are included. The handling quality metrics proposed based on pilot-vehicle systems theory are evaluated against the Neal-Smith flight test data; the agreement is nearly 100 percent. Based upon these results five specific recommendations are offered for the revision of MIL-F-8785B in two areas: the longitudinal dynamic requirements and the flight path stability requirement in power approach. These revisions apply to aircraft incorporating direct lift control and digital flight control mechanizations. Modifications of the criteria proposed for longitudinal dynamics are proposed for the design of advanced display systems. Two additional recommendations are made which apply specifically to the design of digital flight control systems.

A ROLE FOR PILOT-VEHICLE SYSTEMS ANALYSIS

For the conduct of pilot-vehicle systems research or even for the engineering design of an aircraft or FCS, matters of style and personal taste can dictate how one chooses to visualize and model the human pilot's role as an element in the system's dynamics. This is not a satisfactory basis for the development of design specifications for aircraft handling qualities. The rules of engineering conservation must apply, and the community of buyers, manufacturers, and users must all agree on the validity of the specifications to be imposed on the system design. In a practical sense this almost requires that any proposed handling quality specifications be independent of all explicit references to pilot modeling. The viewpoint of these authors is best expressed in the PIO specification based upon the theoretical development of Reference 3. In that work the study of pilot-aircraft system dynamics enabled the derivation of a physical theory for the PIO phenomenon which is independent of the analysis methodology or philosophy from which it is derived. The physical theory for PIO was then translated into a specification for engineering design. The validity of this or any other specification derived from a physical principal can be verified from a dedicated flight test experiment. A physical theory of this sort--once validated with reliable data--is independent of the analysis methods with which it originated.

We believe that handling quality specifications should be independent of pilot-vehicle systems analysis methodology; these are merely means to an end. The tools of analytical handling qualities may be used to understand or correlate data and to aid in the design of experiments; however, their only real use for the support of specification development is to assist ("bootstrap") in the evolution of physical principals on which the handling qualities technology is based.

It cannot be emphasized too greatly that the specification of aircraft design requirements for acceptable handling qualities is an altogether different problem from that of designing an aircraft to have acceptable handling. Any method is acceptable for the design and development of an

aircraft-FCS so long as the results are acceptable. Any design approach will entail a certain number of iterations. Thus, methods for handling qualities prediction such as the Neal-Smith criterion, C* or TRP can all serve a useful function in the design process if they enable the transition of a FCS from the pencil and paper stage to hardware development.

An engineering specification for design acceptability must be right in an absolute sense. It is true that MIL-F-8785B is a design guide of sorts (it all depends upon one's concept of a design guide). But to view it only in those terms is to ignore the reasons for why such specifications exist at all. The intent of MIL-F-8785B is to provide a description of the desired functional performance of the pilot-vehicle system. This, however, is not easily possible to do in any direct quantitative sense without prior identification of a physical, measurable description of handling qualities. There is, as yet, no satisfactory measure for handling qualities other than pilot opinion rating; but that, for reasons that are well known, is not an acceptable metric for use in a design specification.

The philosophy of MIL-F-8785B rests upon the implicit use of pilot opinion rating to "map" airframe and FCS dynamic parameters into regions of acceptable or unacceptable handling qualities. This approach has never been entirely successful; exceptional cases, at both extremes, which violated MIL-F-8785B and its predecessor have always existed. The relationships between handling qualities and modal response parameters of the classic aircraft (ζ_{sp}, ω_{sp}, L_{α}, etc.) have been empirically derived with no substantial guidance from the technology of pilot-vehicle systems analysis. The problem, in essence, is that a reliable method for the prediction or unambiguous correlation of pilot opinion rating (POR) has not existed.

A UNIFIED, ANALYTICAL APPROACH FOR HANDLING QUALITIES SPECIFICATION DEVELOPMENT

The theory of Smith[2] provides the necessary physical and mathematical basis for the prediction of pilot opinion rating. John Arnold's [3] MSE thesis experiment provided the experimental data used by Smith for "calibration" of a pilot rating metric for pitch attitude tracking with a Class IV aircraft

in Flight Phase Category A (fixed base simulation). The result is shown in Slide 2. The handling quality metric $\sigma_{\beta q}$ is hypothesized to represent a measure of pilot effort required for the stabilization of attitude rate q; it is a function of pilot dynamics, airframe and FCS dynamics, display dynamics and threshold (if IFR), and disturbance spectrum.

The estimation or prediction of $\sigma_{\beta q}$ requires an accurate pilot model for the estimation of $\hat{K}_q$ that is not tied too closely to any particular data base for its parameterization. Fortunately, a simpler criterion for handling qualities can be derived from it that is independent of how one models the human pilot.

It is shown in Reference 1 that for all of Arnold's data $\hat{K}_q$ can be treated as a constant value without significant error. This remarkable result suggests that for single-axis, fixed-base tracking of pitch attitude, Cooper-Harper rating is dependent only upon the rms value of q, the airplane's pitching velocity. This dependency is generally confirmed by Arnold's tracking results (Slide 3); quite a lot of scatter is evident at the greater POR. However, it can be seen that the data for each pilot, treated individually, is much more consistent than is the averaged data for all five subject pilots. Slide 4 further illustrates this observation. these data are representative of much of Arnold's data. The remarkable consistency demonstrated between POR and σ_q is more than chance.

Slide 5 shows data taken from Slide 3 for subject number 2. The power curve fit shown resulted from a least-squares fit; it yields an rms error of approximately one unit on the Cooper-Harper scale.

All the POR data for each of Arnold's subjects were fitted in a least-squared error sense with a curve of the form

$$R = K\sigma_q^n \tag{1}$$

The metric K might be termed a "Cooper-Harper" gain. It appears to be approximately constant for each pilot and seems to represent how the individual subject perceives the control task requirements (stability and

performance) against the adjectival descriptors of the Cooper-Harper scale. The results of the least-squares fit are shown in Slide 6. The rms error of fit of Equation (1) and the arithmetic average POR are also tabulated for each subject.

These results indicate that the Arnold data are more consistent on an individual pilot basis than on an average basis across all five pilots. The interpilot differences are systematic and could be interpreted to result from how each pilot interprets the task performance/control effort versus the Cooper-Harper scale.

It is also interesting that the error of fit between the formula $R = K\sigma_q^n$ and the actual pilot opinion rating seems to be strongly correlated with the average rating for each pilot. The connection between $\overline{POR}$ and rms error of fit σ_E can be summarized by the equation

$$\overline{POR} = 5.236 \sqrt{\sigma_E} \tag{2}$$

The rms error of fit of this equation to the data of Table 8 is 0.184 Cooper-Harper units. This relation is construed to represent the expected variation of actual pilot opinion rating from the true nominal rating. The "expected nominal" rating is given by $\overline{POR}$.

If this interpretation is correct, then Equation 2 indicates that increasing task difficulty will result in increased variability in Cooper-Harper ratings. By direct calculation the variation at the Level 1 and 2 boundaries is:

$(POR)_{avg}$	σ_E
3.5	.404
6.5	1.626

We should expect, therefore, that the rms error of Cooper-Harper rating from the true average will be about four times greater at the Level 2 boundary

than at the Level 1 boundary. This accounts for the increased spread in the data of Slide 3 at the greater POR.

The average value of the exponent in Equation 1, for all five pilots, equally weighted, can be computed to be n = 0.497.

In deference to Weber's Square Law, it is assumed that, in general, $n \equiv 0.5$ for all pilots. Thus, the general formula for pilot rating becomes $R = K\sqrt{\sigma_q}$ where K may be a constant but different value for each pilot. The average "rating gain" K across all five pilots for the Arnold experiment is determined to be $K_{avg} = 3.83$. It is therefore concluded that a general rating model for pitch attitude attitude control of Class IV aircraft in Flight Phase Categories A and C is

$$R = 3.83 \sqrt{\sigma_q} \tag{3}$$

POR data from the Neal-Smith flight tests[4] are shown in Slide 7. The ratings of pilot M appear to be biased with respect to those of pilot W. This is indicated by the dashed line. Using the intersection of this bias line with POR = 1 for pilot W as an origin, the two radial lines are drawn to encompass the bulk of the data. The ratio Δ_2/Δ_1 shown is a measure of the relative POR variability at the Level 2 and Level 1 boundaries, respectively. The values used are the maximum of those indicated at each boundary. The result, $\Delta_2/\Delta_1 = 2.3$ is less than the value 4 which we estimated to exist based on the Arnold data. We conclude that the majority of the Neal-Smith data exhibit no more variability than should be expected in any test of this nature.

The 5 "spurious" data points shown in Slide 7 seem to result from interpilot differences in task interpretations--except for Case 6B where turbulence is a likely source. The pilot comments of Reference 4 provide valuable insight into the question of task. Pilot W, in general, seemed to emphasize factors related to air-to-air target acquisition. Pilot M's POR seem to have been based mostly on target tracking after the basic acquisition task had been solved.

It is hoped that the comments made about the dependency between the expected value of pilot opinion rating and the rating variability will be further tested by simulation and flight test. Reference 2 hypothesized that such a relationship exists and that it exists for good physical reasons. The fact is that Cooper-Harper ratings vary because they reflect a physical phenomenon; the variability is systematic. These results indicate that this will be a practical consideration only for the determination of Level 2 boundaries for handling qualities.

Systematic differences between pilots exist. Again, this is no reason to doubt the utility of the Cooper-Harper scale. This effect must be considered when developing a data base or preparing a handling quality specification. The best available and practical solution to this problem at the present time is to use at least two pilot subjects and look for systematic rating differences.

We have proposed that Cooper-Harper rating is proportional to the square root of rms rate error: $R = 3.83 \sqrt{\sigma_q}$ (Slide 8). This result applies for the single axis regulation of pitch attitude with turbulence input in the absence of inertial acceleration cues.

Since the Cooper-Harper scale has a lower bound at 1, this result seems to indicate a contradiction. Note, however, that for $R = 1$, $\sigma_q = .068$ degrees/second = 1 milliradian/second. This value is generally presumed to equal the approximate threshold for visual perception of rate. There is, therefore, no conflict between formula (3) and the Cooper-Harper scale.

It follows that the optimization of handling qualities requires that σ_q approach the value corresponding to the perception threshold; i.e., <u>no tracking is performed</u>, and the Cooper-Harper rating is 1.

THE NO-TRACKING HYPOTHESIS

Optimum handling qualities demands minimum closed-loop control by the pilot.

Application to Specification Development

If the features of open-loop aircraft response to control can be identified which promote pilot tracking, then these imply degraded handling qualities. The quantification of such response properties against available data will then lead in a natural manner to the development of handling quality specifications. This is the procedure that will be implemented in the remainder of this report.

QUANTIFYING THE EFFECTS OF MOTION

The pilot-centered normal acceleration a_{z_p} seems to dominate the effects of inertial motion cues on basic handling qualities. This provided the basis for the PIO theory of Reference 6.

Slide 9 is a typical variation of the parameter $\phi(j\omega)$ with frequency. ϕ is the approximate phase lag of the a_{z_p} loop dynamics (airplane plus pilot). The PIO theory says that if the pitch attitude dynamics (open or closed loop) are highly resonant at frequency ω_R then the susceptibility to PIO will depend upon whether ω_R is greater than or less than the frequency for which $\phi(j\omega)$ is -180°.

The acceleration loop phase margin ϕ_n is, we believe, a useful metric for identification and specification of adverse motion cue effects on handling qulaities--regardless of whether PIO is a potential problem.

For the single loop control of pitch attitude, the behavior of the crossover frequency ω_c with pilot adaptation provides a useful clue for decoding the Neal-Smith data base. The amplitude and phase properties of three stereotype controlled elements are shown on Slide 10. The zeroth order form of required pilot dynamic equalization is indicated for each. These dynamics are representative of a wide range of aircraft dynamics, θ/F_s.

The sensitivity of crossover frequency ω_c to pilot lag T_I for the pure-gain controlled element is indicated in Slide 10. A comparison of ω_c with $\phi(j\omega)$ suggests that the regulation of normal acceleration would demand constant attention and significant precision of control when $\frac{\theta}{F_s}(s) \cong K$ (in violation of the No-Tracking Hypothesis, we note). This would not be true for the other controlled elements. the effect is parameterized by the slope:

$$\frac{d}{d\omega}\left|\frac{\theta}{F_s}(j\omega_c)\right|$$

Note that ω_c must be determined in some manner. The data of Reference 5 support the assumption that, to a good approximation, ω_c is parameterized by the slope parameter. Data from Reference 5 are shown in Slide 11. Note that the relation is an implicit one. This method for the estimation of the criterion frequency ω_c is assumed throughout the remainder of this presentation.

Slide 12 summarizes the four principal handling quality metrics that we have investigated. The time-to-first peak of q(t) was selected to discriminate important features of aircraft response during large amplitude maneuvering control that might escape our notice in a linear tracking experiment. The effects of sign convention on $\frac{\theta}{F_s}$ are removed by the division by $M_{\delta e}$.

QUANTIFYING THE METRICS

The Neal-Smith data were used to test the applicability of the above four metrics and to quantify their effect on POR.

Slide 13 indicates that tq = 0.2 seconds is a lower limit. This is reasonable, based on human time delay estimated from closed loop tracking (generally near 0.2). When tq < 0.2, the response is "too abrupt" and, in effect, the pilot feels forced to track the results of his previous input--the No-Tracking Hypothesis again. It is possible that no upper limit should be placed on tq since large tq results from excessive phase lag. However,

we have tentatively specified $t_q \leq 0.9$ as being consistent with most of the data.

Slide 14 shows a plot of pitch attitude phase (airplane including FCS) vs POR averaged for both pilots for each case tested. The results appear to be chaotic. Careful inspection, however, will reveal that the data may be generally grouped according to the slope parameter. The nominal bounds are shown for slope ≥ -2 decibels/octave (note that this is the alegebraic sense).

When those data cases for which $\frac{d}{d\omega}\left|\frac{\theta}{F_s}\right| \geq -2$ or which violate $0.2 \leq t_q \leq 0.9$ are removed, the results are as shown in Slide 15. The boundaries we believe, reflect the nonlinear nature of the Cooper-Harper scale. That is, the discontinuities in slope belong there until such time as a linear scale can be devised. The two "spurious" data points 7F (pilot W) and 1A can both be rationalized. Case 7F is especially interesting since grossly different POR were obtained for this case for pilot M (also shown). Both points 1A and 7F-W can be moved inside the boundaries when the criterion frequency ω_c is increased. As an a priori prediction, however, the combination of the three parameters t_q, $\frac{d}{d\omega}\left|\frac{\theta}{F_s}\right|$ and $\measuredangle \frac{1}{M_{\delta e}} \frac{\theta}{F_s}$ yields 94 percent success.

The appropriate regions of level 1/2 handling qualities are indicated on Slide 15.

For the same cases, Slide 16 shows the variation of $\phi(j\omega_c)$ with pilot induced oscillation rating (PIOR). All except five cases exhibit a very consistent trend. These five, however, appear to be cases that would violate the proposed PIO specification (developed from Reference 6). All five cases would be correctly categorized by the boundaries shown in Slide 16 if the criterion frequency were increased to simulate a "more aggressive" piloting technique. Note that case 7F (pilot W) is one of the errant cases. The spurious cases are discussed in more detail in Reference 1.

A DIRECT LIFT CONTROL CRITERION

Slide 16 omits those cases for which the slope criterion is violated; i.e., $\frac{d}{d\omega}\left|\frac{\theta}{F_s}\right| \geq -2$ db/oct. This was done on the presumption that such cases are those for which essential handling quality problems exist due to adverse normal acceleration cue effects. If we plot $\phi(j\omega_c)$ vs PIOR for <u>only</u> those aircraft-FCS cases which violate only the slope criterion (i.e., for which $\frac{d}{d\omega}\left|\frac{\theta}{F_s}\right| \geq -2$ db/oct) then the resulting trend will hopefully apply regardless of pitch dynamics. In particular, we suspect it will apply when a DLC system is employed--for which θ may be automatically controlled, for example. For these cases $\phi(j\omega_c)$ is plotted vs PIOR in Slide 17. A strong correlation exists. Note that there are no Level 1 cases. The Level 1 boundary is estimated by a slight extrapolation.

COMMENT ON THE PIOR SCALE

It has been suggested that the PIOR scale is redundant and that it be discarded. We do not believe this. The scale is highly compressed, truncated, and therefore nonlinear--this was seen in Slides 16 and 17.

We believe that the PIOR scale, as it has been used, quantifies the adverse effects of normal acceleration on handling qualities. It complements the Cooper-Harper scale which emphasizes the ability of a pilot to perform a control task without explicit consideration of motion cue effects. For advanced FCS with direct force control modes, the PIOR scale may prove valuable.

SUMMARY OF PROPOSED SHORT-PERIOD CRITERIA

The results of previous slides are summarized in Slide 19. It should be noted that these criteria would seem to apply equally well to advanced display design. When a display is used for a particular task (fire control, energy maneuvering, etc.) such that the pilot must track the error in a single cue x, then substitute x for r, $\mathring{x}$ for q and discard the motion

criterion $\phi(j\omega_c)$. The result is a set of criteria for the preliminary design of display systems.

FLIGHT PATH STABILITY

A new criterion for flight path stability in power approach can be developed in a manner completely analogous to that for attitude control. The essentials are listed on Slide 20.

The time response measure t_γ was defined as the time to 90 percent peak because of the near-aperiodic nature of the $\dot{\gamma}$ responses for many aircraft. The simplified treatment used for addressing the difficult question of control technique effects in power approach is fully justified in Reference 1. Here, we note that when an aircraft-FCS demands a level of control complexity significantly greater than the simple use of F_s or δ_t to correct γ errors, then the pilot is forced to track; by the No-Tracking Hypothesis, the handling qualities will be degraded.

The peculiar damping parameter D is defined in the manner indicated in Slide 20 because it is a "pilot-centered" perspective of flight path stability and is probably more flight testable than a more conventional, linear measure of response damping.

The proposed criteria for flight path stability are tabulated in Slide 21. These results were empirically determined from an analysis of 17 aircraft. Slide 22 is a comparison between flight experience (as best as can be estimated from personal experience or from published data sources) and the criteria proposed. The agreement is complete except at the Level 1 boundary where some minor uncertainty occurs.

Observe that the NATOPS-prescribed carrier approach task falls into the "Throttle" column. The F-111B is seen to be much more poorly rated for the carrier approach task than it is predicted to be for field landing (the USAF approach seems to, more often than not, emphasize a "point the nose at the end of the runway and come in hot" technique). Both predictions concur with

USN and USAF experiences. Conversely, the F-4B gets rave reviews for carrier approach but is a more difficult airplane when flown with a technique that emphasizes elevator control of flight path angle.

The effect on handling qualities of the F-4 of direct lift control are also indicated. The DLC airplane is shown in the elevator column. This assessment is supported with flight test experience with this airplane. One problem with DLC that must be solved is the design of a cockpit manipulator. We suggest that an attitude-hold FCS would be appropriate with stick (center or side) used for DLC command. Thumbwheels on top of control sticks, with all systems integration being done within the pilot's skull hardly seems like the way to fly in the computer age.

DFCS REQUIREMENTS

Slide 23 shows some specific requirements proposed in Reference 1 for digital FCS. Time does not permit a discussion of these here beyond noting that DFCS hardware choices can have a potentially significant effect on handling qualities. This is particularly true of A-D and D-A conversion techniques. The handling qualities impact must be carefully evaluated for each specific system before the hardware design becomes immutably frozen.

OTHER TOPICS

Slides 24 and 25 indicate the maximum tolerable values of short-period damping ratio and equivalent systems time delay for Level 1 handling qualities. These results were derived by using the proposed phase criterion

$$\measuredangle \frac{1}{M_{\delta e}} \frac{\theta}{F_s}(j\omega_c) \geq -130°$$

and solving for τ_E to give a phase of -130° for the cases shown.

The point of this exercise is to indicate that the phase criterion seems to be quite powerful, easily applied, and yields results consistent

with other evidence. For instance, the $(\zeta_{sp})_{max}$ values are in reasonable agreement with MIL-F-8785B; based on these results, the Level 1 limit of the specification seems to be conservative as Mayhew has suggested (Proposed Revisions to MIL-F-8785B). Various investigators have suggested that $\tau_E = 0.1$ might be used as a handling qualities boundary. The examples of Slide 25 support that to a certain extent, but suggest that such a requirement is too simplistic and restrictive.

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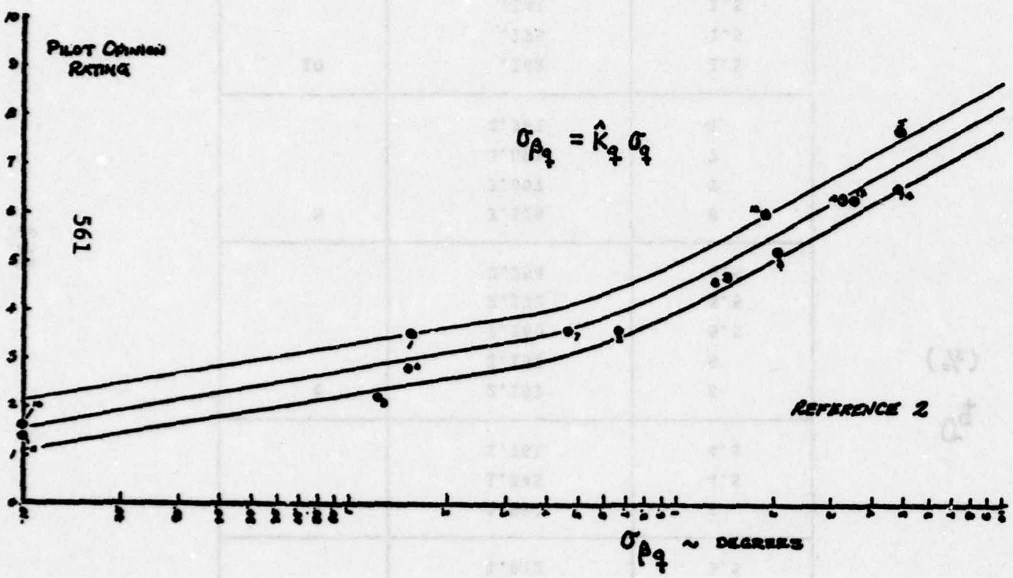
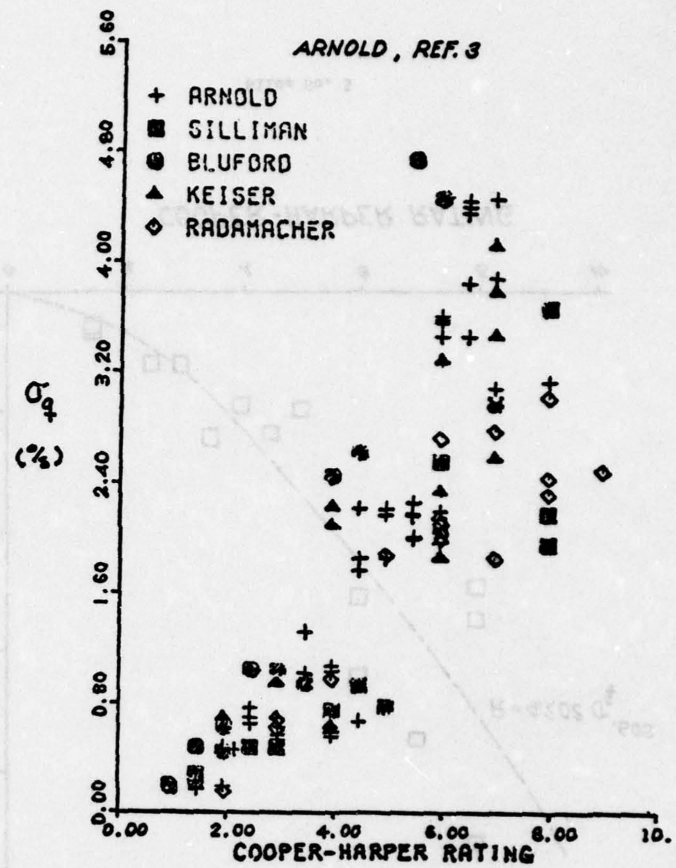
Slide No. 1



Slide No. 2



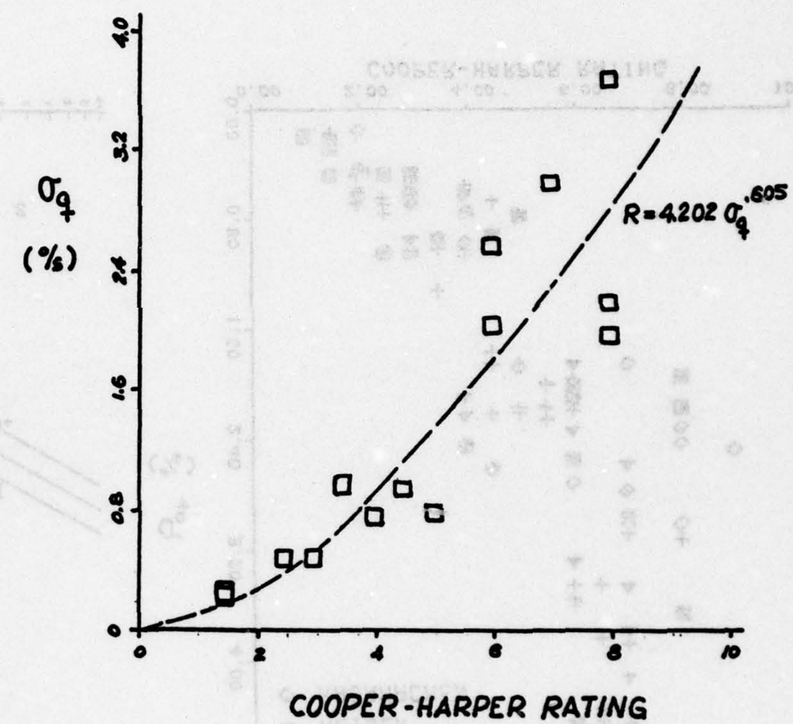
Slide No. 3

SUBJECT NO. 5

CASE	RMS PITCH RATE °/s	POR
2	1.035	4
	1.015	3.5
	1.012	3.5
	1.012	3.5
3	1.852	5
	1.845	4.5
	1.761	4.5
4	2.193	6
	2.152	6
	2.160	5.5
	2.172	5.5
	2.254	5.5
5	3.129	8
	3.007	7
	3.081	7
	2.962	7
10	.162	1.5
	.175	1.5
	.141	1.5
	.168	2
	.171	2

Slide No. 4

SUBJECT 2

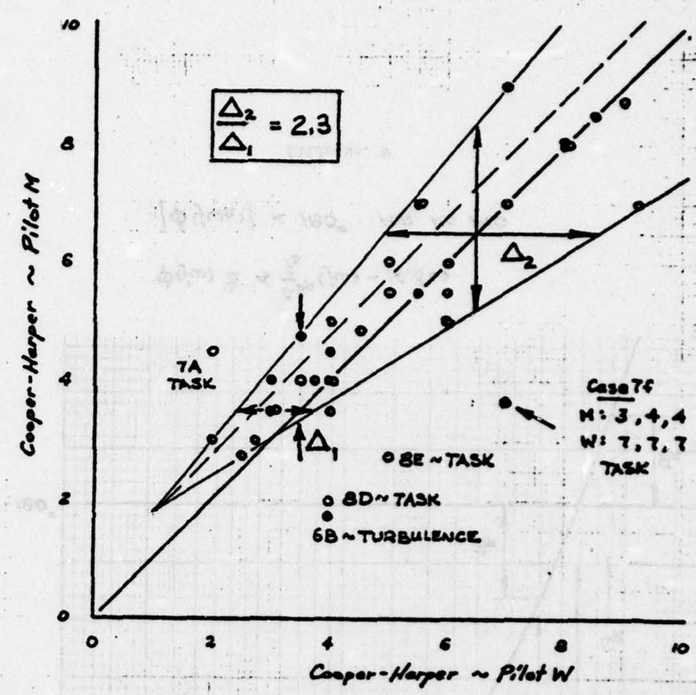


Slide No. 5

$R = K\sigma_q^n$				
SUBJECT	K^3	n	σ_ϵ'	$\overline{POR}^2$
BLUFORD	2.606	.527	.218	2.6
ARNOLD	3.559	.450	.588	4.4
KEISER	3.844	.431	.906	5.1
SILLIMAN	4.202	.605	.983	4.9
RADAMACHER	4.391	.474	.982	5.2
$R = 3.83\sqrt{\sigma_q}$				
$\overline{POR} = 5.236\sqrt{\sigma_\epsilon}$				

1. σ_ϵ = RMS (POR − R)
2. $\overline{POR}$ = ARITHMETIC AVERAGE (EACH SUBJECT)
3. K = COOPER-HARPER GAIN
4. $\dfrac{(\sigma_{POR})_{6.5}}{(\sigma_{POR})_{3.5}} \cong 4$

Slide No. 6



Slide No. 7

THE NO-TRACKING HYPOTHESIS

$$R = 3.83\sqrt{\sigma_q}$$

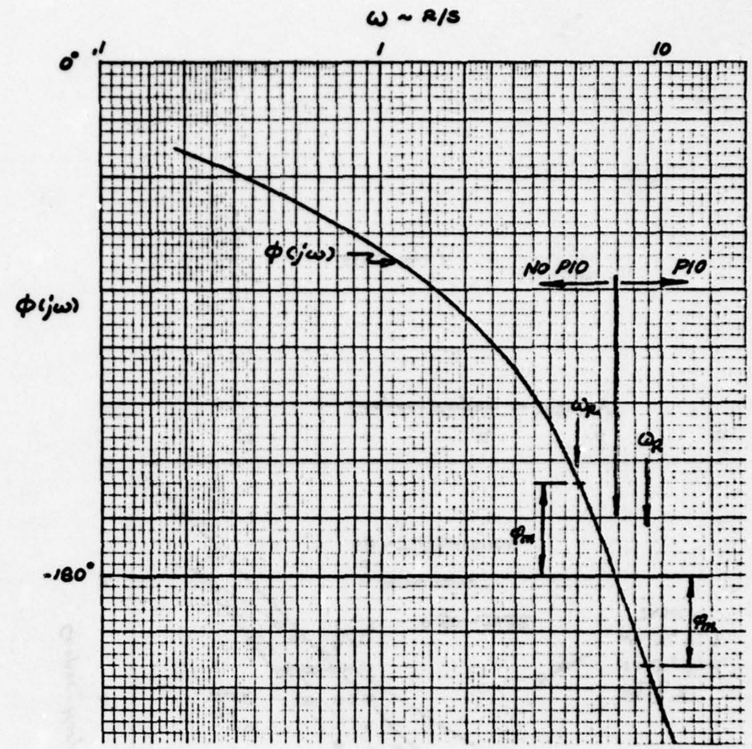
$\sigma_q \rightarrow 1$ mR/s $\Rightarrow$ $R \rightarrow 1$

OPTIMUM HANDLING QUALITIES REQUIRES
NO UNNECESSARY TRACKING

THE SPEC PROBLEM:

IDENTIFY FEATURES OF AIRCRAFT-FCS RESPONSE WHICH FORCE A PILOT TO TRACK

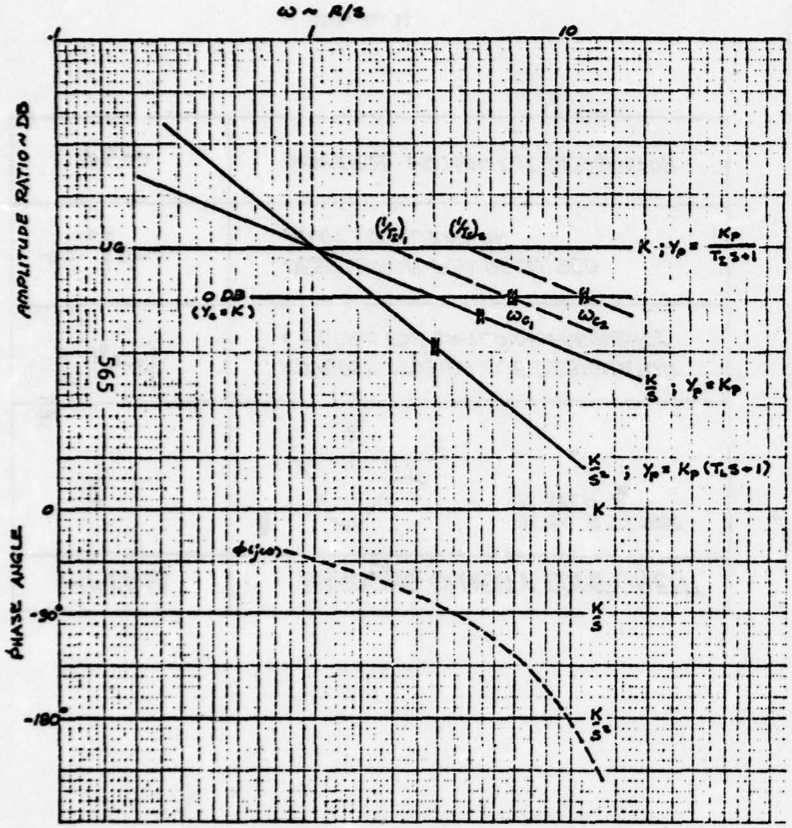
Slide No. 8



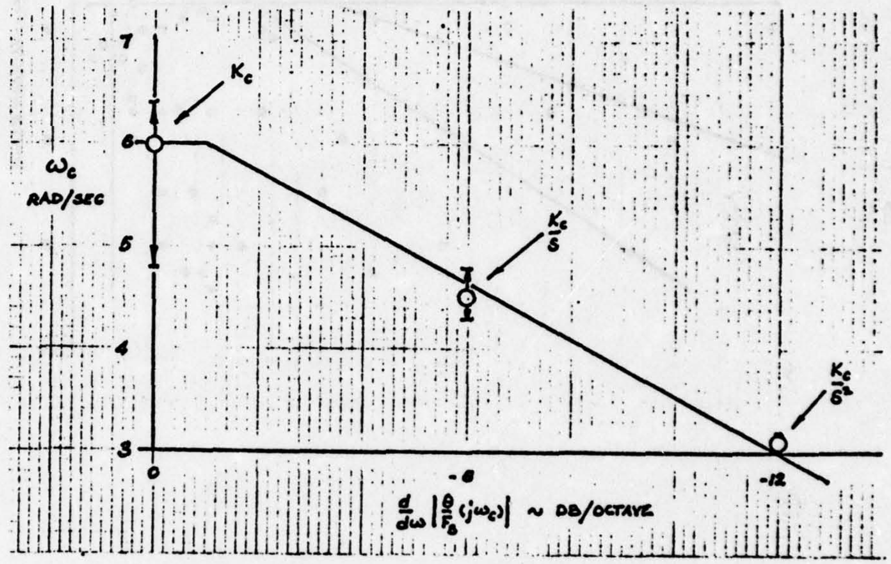
$\phi(j\omega) \triangleq \measuredangle \frac{a_{z_p}}{F_s}(j\omega) - 14.3\omega$

$|\phi(j\omega_R)| < 180°$ FOR NO PIO

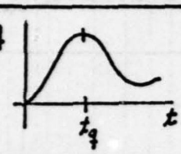
Slide No. 9



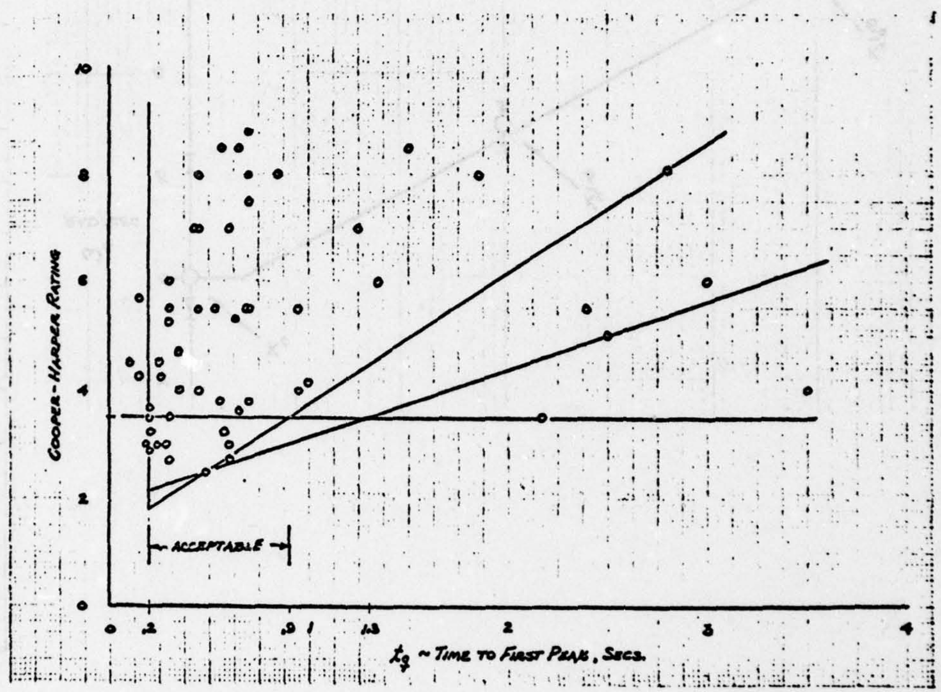
Slide No. 10



Slide No. 11

METRIC	HANDLING QUALITY COMPONENT
t_q	 INITIAL RESPONSE TO STEP F_s
$\frac{d}{d\omega}\left\lvert\frac{\theta}{F_s}(j\omega_c)\right\rvert$	ADVERSE MOTION CUE DESIGNATOR & METRIC FOR CRITERION FREQUENCY
$\angle \frac{1}{M_{\delta e}}\frac{\theta}{F_s}(j\omega_c)$	DESIGNATOR FOR REQUIRED PILOT EQUALIZATION
$\phi(j\omega_c)$	ADVERSE MOTION CUE DESIGNATOR

Slide No. 12



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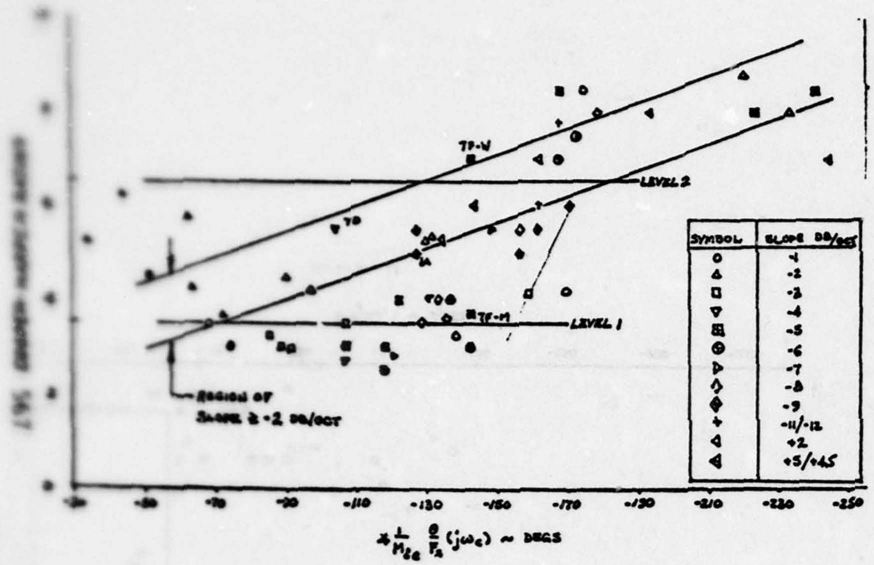
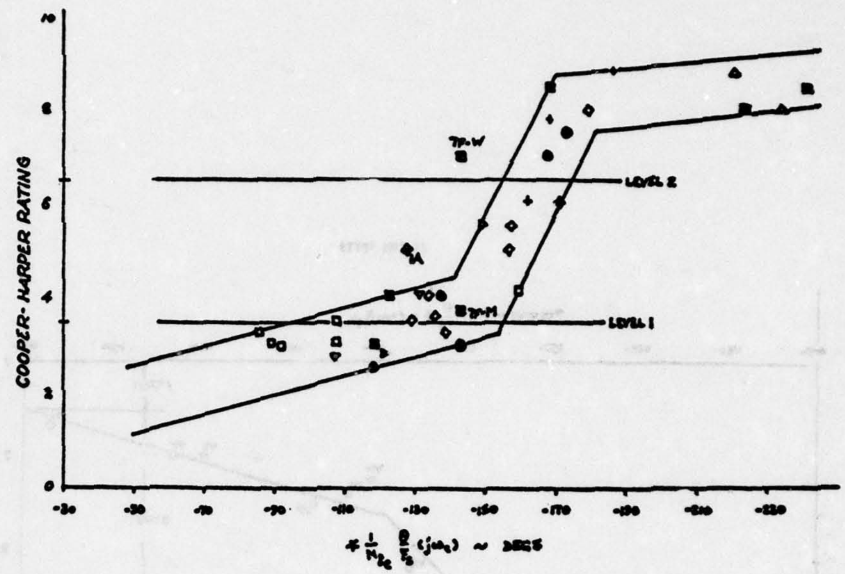
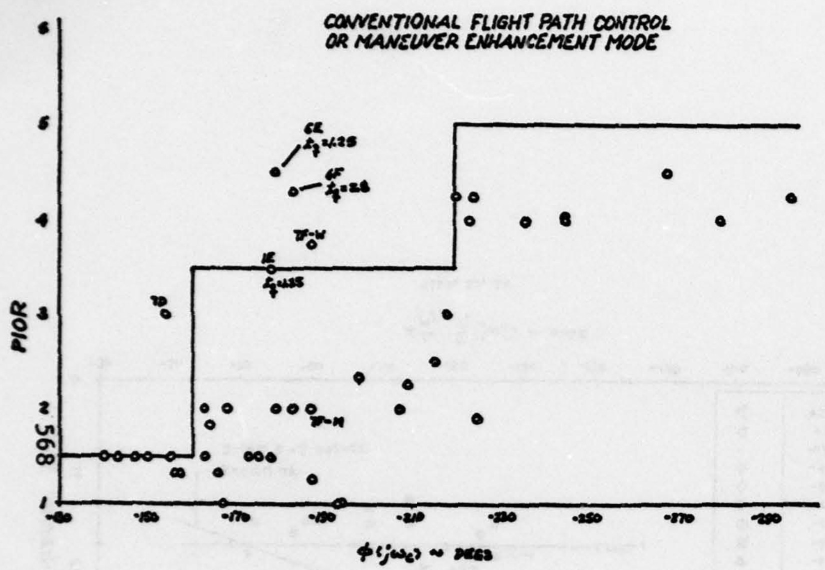


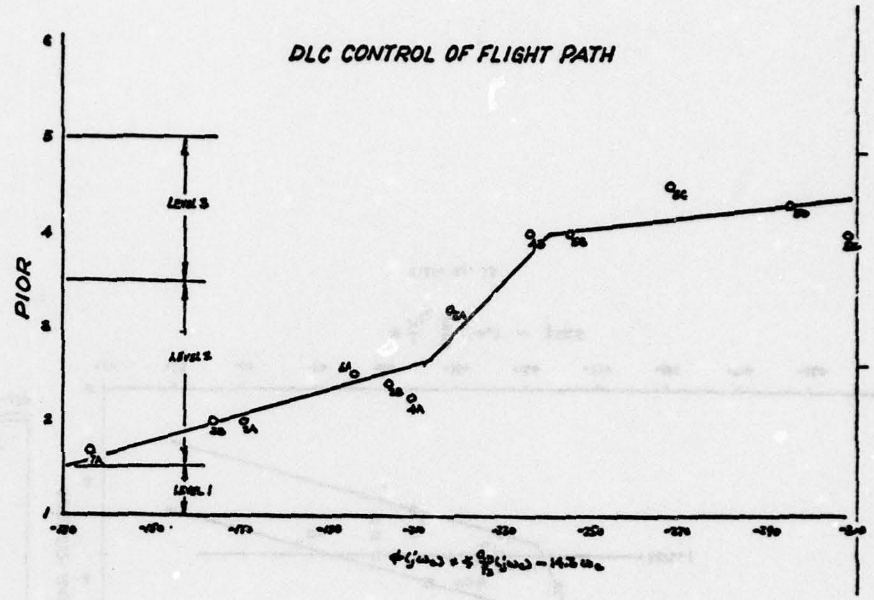
Slide No. 14



Slide No. 15



Slide No. 16.



Slide No. 17

THE PIOR SCALE

- HIGLY NONLINEAR -- COMPRESSED
- QUANTIFIES EFFECTS OF NORMAL ACCELERATION ON HANDLING QUALITIES
- COMPLEMENTS COOPER-HARPER
- SHOULD BE REVISED & USED--NOT SCRAPPED

Slide No. 18

PROPOSED SHORT-PERIOD CRITERIA (SUMMARY)

(1)	$0.2 \le t_q \le 0.9$	ALL LEVELS
(2)	$\frac{d}{d\omega}\left\|\frac{\theta}{F_S}(j\omega_c)\right\| < -2$ DB/OCT	LEVELS 1 & 2
(3)	$\measuredangle \frac{1}{M_{\delta e}}\frac{\theta}{F_S}(j\omega_c) \ge -130^\circ$	LEVEL 1
	$-130^\circ > \measuredangle \frac{1}{M_{\delta e}}\frac{\theta}{F_S}(j\omega_c) \ge -170^\circ$	LEVEL 2
	$-170^\circ > \measuredangle \frac{1}{M_{\delta e}}\frac{\theta}{F_S}(j\omega_c)$	LEVEL 3
(4)	$\phi(j\omega_c) \ge -160^\circ$	LEVEL 1
	$-160^\circ > \phi(j\omega_c) \ge -220^\circ$	LEVEL 2
	$-220^\circ > \phi(j\omega_c)$	LEVEL 3

FOR CONTROL-DISPLAY INTEGRATION:

(1) SUBSTITUTE: x FOR θ, $\dot{x}$ FOR q

(2) DISCARD CRITERION #4

Slide No. 19

FLIGHT PATH STABILITY

$\dot{\gamma}$ ANALOGOUS TO q

$t_{\dot{\gamma}}$ " " t_q

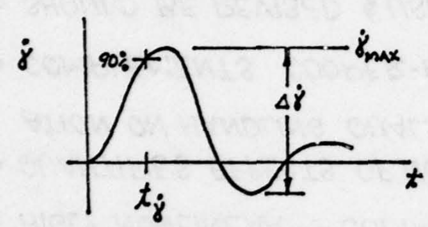
CONTROL TECHNIQUE :

(1) STEP F_s @ U_o = CONSTANT

(2) STEP δ_T @ θ = CONSTANT

INVOKE THE NO-TRACKING HYPOTHESIS

DEFINITIONS :



$D \triangleq \Delta\dot{\gamma}/\dot{\gamma}_{MAX}$ .... "DAMPING"

Slide No. 20

	AUTHORITY	TIMELINESS	PRECISION
	$\dot{\gamma}_{MAX}$	$t_{\dot{\gamma}}$	D
LEVEL 1	.4	3.5	1.5
LEVEL 2	.2	6.0	2.0
LEVEL 3	.1	NA	3.0

FLIGHT PATH STABILITY CRITERIA (PROPOSED)

Slide No. 21

DATA COMPARISON : POWER APPROACH

PROPOSED CRITERIA

	CONTROL TECHNIQUE		
	ELEVATOR	THROTTLE	FLT. EXPERIENCE
LEVEL 1	F-8 DLC P-3B T-2C DC-8 A-3B YF-16 NT-33 F-111 A A-6E	F-4B F-5A/E/F T-2C A-6E	P-3B F-4B (USN) F-8 DLC T-2C YF-16 F-111A (AF) A-6E DC-8 F-5A/E/F F-4C (AF) N-T33 A-3B
	C-5 RA-5C	A-3B	C-5
LEVEL 2	F-8D F-5A F-4B	YF-16 F-111A C-5 RA-5C DC-8 P-3B F-8D F-4M	RA-5C F-111A (USN) F-8 F-4M
LEVEL 3			

Slide No. 22

DFCS REQUIREMENTS

FRAME RATE :

LOW FREQUENCY LIMIT $\ast \frac{1}{M_{\delta_e}} \frac{\theta}{F_s}(j\omega_c)$

HIGH FREQUENCY LIMIT $f < \frac{\theta_b \omega_c}{10 R/2^n}$, Hz

CONTROL ROUGHNESS :

$$\left|\frac{q}{\delta_e}(j\omega_c)\right| \Delta\delta_e \leq .001 \text{ R/S}$$

$$\left|\frac{a_{z_P}}{\delta_e}(j\omega_c)\right| \Delta\delta_e \leq .01 \text{ G}$$

Slide No. 23

MAXIMUM ζ_{SP}
(CLASSIC AIRPLANE)

$\frac{1}{T_{\theta_2}}$	ω_{SP}	$(\zeta_{SP})_{MAX}$
0.5	2.0	2.1
1.5	6.0	2.1
2.0	9.0	2.5

Slide No. 24

EQUIVALENT SYSTEM:
TIME DELAY

$$\frac{\theta}{F_s}(s) = \frac{K\,(s + 1/T_E)\; e^{-\tau_E s}}{s\,(s^2 + 2\zeta_E \omega_E s + \omega_E^2)}$$

$\zeta_E = 0.5\ ,\ 1/T_E = 1.$

ω_E	$(\tau_E)_{MAX}$
2	-.042
4	.009
6	.089
9	.205

Slide No. 25

Dwight Schaeffer, Boeing: What was the basis for your expression

$$R = 3.83\sqrt{\sigma q} \ ?$$

Answer: The simple formula $R = 3.83 \sqrt{\sigma q}$ was derived from data for fixed-base simulation of pitch tracking in turbulence. The formula appears not to address task performance but, in fact, does since q & θ are linearly related. If the task requirements were completely changed, this would be reflected by a change of gain constant. In general, for pitch tracking with no adverse effects of normal acceleration, $R = K\sqrt{\sigma q}$ seems to reasonably approximate the relation between σq & Cooper-Harper ratings. I have termed K the "Cooper-Harper" gain. Based on Arnold's data K is a constant of the individual pilot and seems to establish how the individual calibrates his task performance and the degree of difficulty with the adjectives of the Cooper-Harper scale. The rating formula is not proposed for applications; it was used by me to identify q(t) as the essential response of importance to handling qualities and to develop the No Tracking Hypothesis as a philosophy to guide the search for handling quality parameters.

Bill Rickard, Douglas: In the Cooper-Harper rating versus phase criterion, at 130° what is the rating?

Answer: The boundaries shown merely illustrate the tightness of fit between the phase angle criterion and the data. As a specification, I suggested the mid-point of the data lying along the 3.5 rating line. In the region of the "knee" of the data, a small change in pilot technique may yield a large change in pilot opinion rating. This is suggested by the correlation shown and constitutes what Rogers Smith has termed a "handling qualities cliff".

CLOSED LOOP FLYING QUALITIES CRITERIA

Frank L. George

AFFDL Symposium and Workshop on
Flying Qualities and MIL-F-8785B

12-15 September 1978

Air Force Flight Dynamics Laboratory
Air Force Wright Aeronautical Laboratories
Air Force Systems Command
Wright-Patterson AFB, Ohio

CLOSED LOOP FLYING QUALITIES CRITERIA

SUMMARY

This paper addresses the subject of closed loop, or pilot-in-the-loop, flying qualities requirements for military airplanes. The need for such criteria and their potential advantages are discussed. A definition of the term closed loop criteria, as used in this paper, is given. Various methods for developing closed loop criteria are reviewed and examples of such criteria are discussed. The strengths and weaknesses of this approach to flying qualities criteria are pointed out along with potential research areas.

THE NEED FOR CLOSED LOOP CRITERIA

Consider the following trends in military aircraft, particularly tactical and strategic vehicles:

large flight/performance envelopes
highly complex systems
high cost systems

These factors dictate that the military service receive the greatest possible mission effectiveness and survivability from these piloted aircraft systems.

Because of their large operating envelopes, these military airplanes must perform efficiently and maintain a good pilot-airplane interface over a wide variety of flight conditions. Current design methods and anticipated mechanization concepts to achieve these goals lead to configurations that include complex stability and control augmentation as an essential element. The resulting dynamic system, shown in Figure 1, may exhibit significantly different responses to various stimuli such as external disturbances, mode changes or failures, as opposed to deliberate commands.

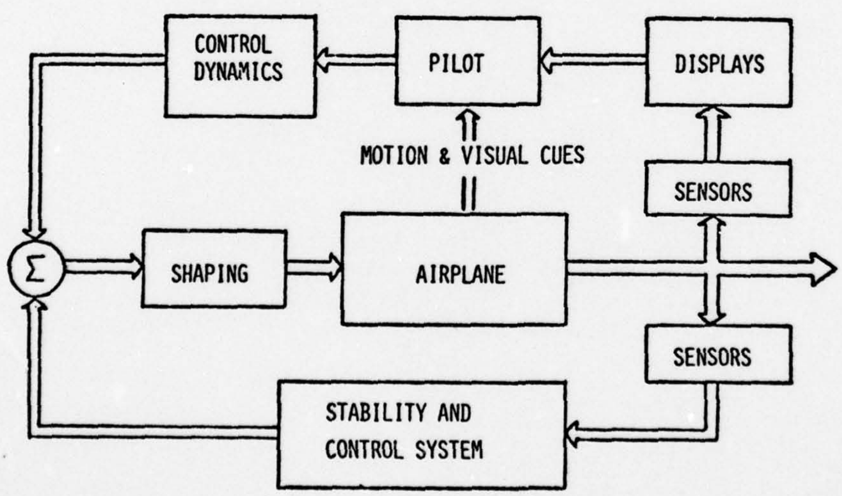


FIGURE 1. PILOT-AIRPLANE CLOSED LOOP SYSTEM

Note that Figure 1 should be viewed as a generic system diagram where, for example, the "stability and control" system block may be a complex subsystem containing automatic control functions as well as stability augmentation.

In addition to the variable dynamics described above, it is desirable to tailor the system dynamic characteristics to enhance the performance of various mission tasks. Thus, for the reasons summarized below, it becomes essential to have closed loop, or pilot-in-the-loop, flying qualities criteria:

- potentially large differences between open loop and closed loop dynamics

- desire to be consistent with military emphasis on improved mission effectiveness

- desire to be consistent with the design philosophy of task-oriented multimode controls.

A DEFINITION OF CLOSED LOOP FLYING QUALITIES

As discussed in the previous section, closed loop flying qualities deals with the dynamic characteristics of the total pilot-in-the-loop system of Figure 1. This certainly is not a new idea. For example, Reference 1 says much the same thing in contrasting open loop and closed loop flying qualities as shown in Figure 2.

OPEN LOOP:

- CHARACTERISTICS RELATED TO UNATTENDED OPERATION - PREDOMINANTLY THE STABILITY AND RESPONSE TO STIMULI OF THE [AUGMENTED] AIRCRAFT ALONE

- CHARACTERISTICS RELATED TO THE BASIC ABILITY OF THE AIRCRAFT TO EXECUTE MISSION-RELATED OR EMERGENCY MANEUVERS ASSUMING AN IDEAL PROGRAMMED CONTROLLER - PREDOMINANTLY THE LIMITING (MAXIMUM) CONTROL AND STEADY-STATE RESPONSE CHARACTERISTICS

CLOSED LOOP:

- CHARACTERISTICS RELATED TO CLOSED LOOP CONTROL - PRIMARILY THOSE DYNAMICS INVOLVED WITH THE PILOT/AIRCRAFT INTERACTION IN A FEEDBACK CONTROL SITUATION

FIGURE 2. OPEN LOOP vs CLOSED LOOP FLYING QUALITIES

Hence, the subject of this paper can be defined in the following deceptively simple-appearing statement.

> Closed loop flying qualities consist of those combined pilot-vehicle system characteristics that impact the satisfactory performance of precise pilot-in-the-loop tasks.

The true complexity of this subject is reflected by the considerable amount of research, and technical documentation which addresses the topic. References 1 through 6 are a small sample of the pilot-in-the-loop flying quality theories that have been proposed.

CLOSED LOOP FLYING QUALITIES CRITERIA OBJECTIVES

As with the definition above, the fundamental goal of closed loop flying qualities analysis and requirements is deceptively easy to state. That is:

> We must develop the capability to understand what the pilot needs to know and what he needs to do in order to complete a task - and relate that knowledge in some quantitative sense to system design parameters.

Figure 3 serves to illustrate this goal as well as some of the complex interactions and dynamic effects that complicate the situation.

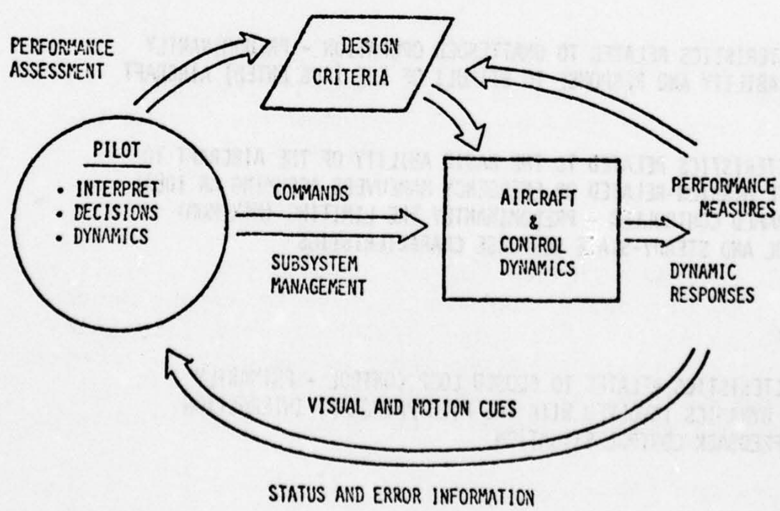


FIGURE 3. CLOSED LOOP FLYING QUALITIES ANALYSIS PROBLEM

For example, the pilot is recognized to be a complex dynamic element of the closed loop system, and considerable attention has been given to analytically modeling his dynamics when acting as a tracking control element. References 7-10 describe the two most widely used approaches to modeling and analysis of the pilot as a control element. However, it is necessary to understand the pilot's total role in the closed loop system to develop adequate flying qualities criteria. This broader understanding, and associated modeling capability, is needed to help identify appropriate closed loop flying qualities metrics.

One example of broadening our understanding in the closed loop situation deals with the concept of control harmony. In the present requirements, harmony deals primarily with inter-axis consonance of controller characteristics. In the context of closed loop flying qualities criteria, the concept of harmony can be expanded to encompass the pilot's overall interface with the aircraft systems. This interface exists whether the pilot is performing as an active controller or as a monitor. Thus the pilot needs appropriate displays and cues for both actual and anticipated control actions.

It therefore becomes necessary to consider the pilot's anticipated role in the closed loop system when defining appropriate closed loop criteria. For example, if closed loop criteria are to be defined in terms of task performance measures, it must be recognized that significant differences in task performance can occur between manual and automatic control modes because of different parameters and control paths that may be employed. Hence, the primary and backup operating modes must be determined, and appropriate criteria provided for each.

To summarize, if the pilot and aircraft system are considered as coupled dynamic elements, then it is also appropriate to treat the design criteria as being dynamic in the sense that they should be tailored to the particular application. The nature of closed loop metrics and the form of criteria consistent with this philosophy are the topics of the next section.

APPROACHES TO CLOSED LOOP CRITERIA

As a first step in developing new closed loop criteria, it is logical to review the present requirements, both to assess needs and to establish a baseline. MIL-F-8785B, Reference 11, contains some 31 paragraphs or subparagraphs in its four major requirements sections that could be construed to address closed loop flying qualities. These requirements address the following general areas:

control forces and feel in maneuvering flight,
nonlinearities and transient effects,
pilot-control-system dynamic stability (e.g. pilot induced oscillations).

The approach in these current requirements is to define airplane characteristics that will lead to an appropriate pilot interface. (An example is controller force and feel characteristics). One potential approach to future closed loop criteria would be to continue within the present framework. That is, establish numerical bounds on the airplane system characteristics, those already defined plus appropriate new ones, perhaps using an expanded task/flight phase grid structure. An advantage to this approach is, of course, that it builds on the familiarity of the present criteria. A disadvantage, however, is the reliance of many of these criteria on empirical data to establish numerical bounds. This reliance on empirical data limits the utility of the criteria in evaluating new designs that fall outside the current data base; it also makes criteria revision a potentially expensive and lengthy process due to the need for generating and validating new data. However, a more fundamental limitation of this approach is that the criteria do not directly address the effectiveness, or task performance, of the coupled pilot-in-the-loop system.

Most alternatives to the approach discussed above involve in some way the topics of pilot modeling and pilot-vehicle analysis. For example, criteria could be defined in terms of closed loop task performance such as landing accuracy or target tracking error. A design engineer would quite likely use some form of pilot-vehicle analysis for at least preliminary evaluation of proposed configurations against such criteria. An approach to criteria that involves pilot modeling more explicitly is to establish bounds on pilot control activity (including workload) to achieve desired closed loop task performance characteristics. The obvious question regarding these and similar approaches centers on the understanding of the pilot's function in the closed loop system and the ability to model his essential characteristics. As anyone involved with flying qualities knows, this question has been addressed in research and debate for many years. Two methods of analytically modeling the pilot have evolved that have been applied in a variety of situations with reasonable success. Figure 4 outlines the basic features of these two methods. References 12, 13, and 7 trace the evolution of the servo model while References 8,9, and 10 describe development of the optimal control model.

While the two pilot model approaches shown in Figure 4 look different, the fundamental assumptions and limitations are the same. That is, both assume the pilot acts in some optimal fashion and can be represented by a linear model. Also, both are limited in application - to closed loop, small amplitude tracking tasks. Within these limits, however, the models have been used with some success to analyze closed loop flying qualities problems and derive criteria. Examples of pilot-vehicle analysis applied to criteria development include Neal and Smith's pitch dynamic criterion described in Reference 14, the turn coordination requirement developed by Ashkenas, et. al. in Reference 15, and the pilot induced oscillation requirement proposed by Smith in Reference 16.

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PROCEEDINGS OF AFFDL FLYING QUALITIES SYMPOSIUM HELD AT WRIGHT --ETC(U)
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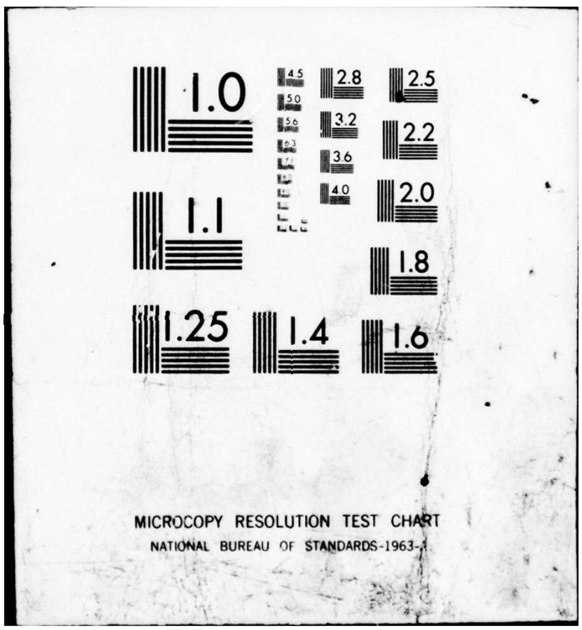
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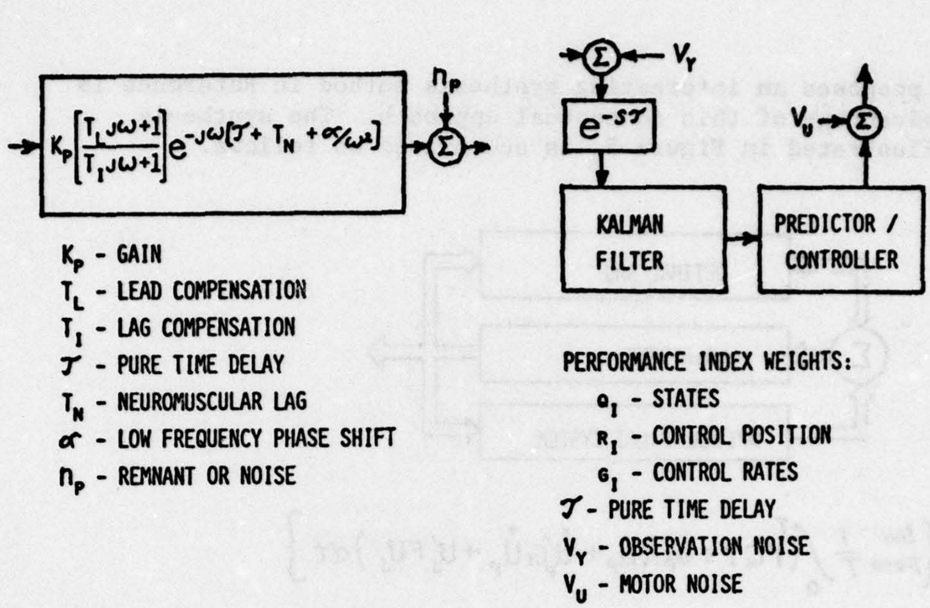


A) SERVO MODEL

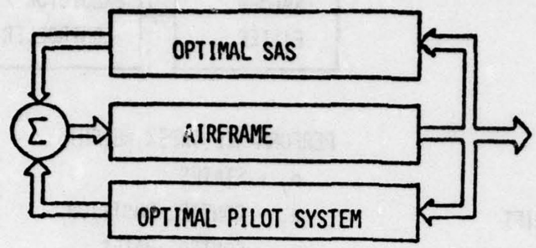
B) OPTIMAL CONTROL MODEL

FIGURE 4. PILOT MODELING APPROACHES

The common element of these three developments was the approach of defining criteria to achieve desirable closed loop response characteristics directly in terms of measurements of the system dynamic characteristics. The two following examples illustrate the potential of defining criteria in terms of closed loop task performance, as correlated with pilot opinion rating.

References 5 and 17 propose methods for analytically predicting Cooper-Harper Pilot Opinion Rating using pilot modeling techniques. Both references rely on the optimal control model, but propose different metrics to correlate the closed loop system characteristics with pilot rating. The metric in Reference 5 derives from the pilot model structure, while Reference 17 uses the optimal control performance index which includes both pilot control and closed loop response characteristics. It is generally accepted that the pilot's subjective Cooper-Harper rating of flying qualities is a function primarily of his workload (e.g. control activity) and task performance (e.g. observed closed loop errors). Hence, this approach admits the possibility of defining closed loop criteria very simply in terms of the pilot's assessment of task performance. Since Reference 18 has defined a relationship between Cooper-Harper rating and levels of flying qualities, this approach presents a viable alternative for evaluating pilot-vehicle system characteristics in those cases where other forms of definitive criteria are lacking, as discussed below.

Schmidt proposes an interesting synthesis method in Reference 19 which takes advantage of this conceptual approach. The synthesis procedure, illustrated in Figure 5, is summarized as follows.



$$J = E\left\{ \lim_{T\to\infty} \frac{1}{T} \int_0^T \left(Y'QY + U_p'RU_p + \dot{U}_p'G\dot{U}_p + U_s'FU_s \right) dt \right\}$$

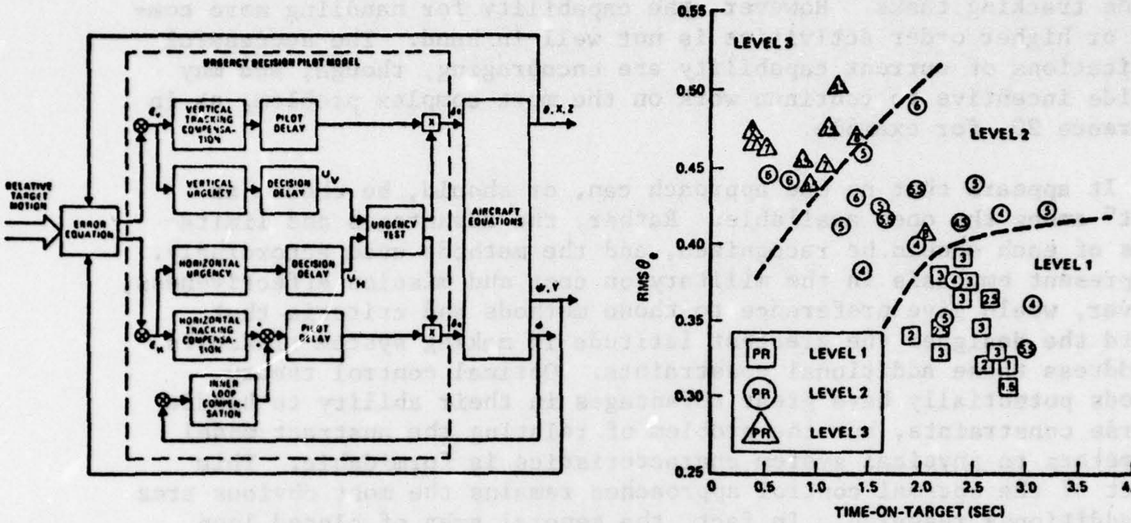
$$\left.\begin{array}{l} 1)\quad A_p\Sigma + \Sigma A_p + W - \Sigma C_p' V_y^{-1} C_p \Sigma = 0 \\ 2)\quad -\begin{bmatrix} A & B \\ -G^{-1}K_{p_3} & -G^{-1}K_{p_4} \end{bmatrix}' K_s - K_s' \begin{bmatrix} A & B \\ -G^{-1}K_{p_3} & -G^{-1}K_{p_4} \end{bmatrix} - P \\ \qquad + K_s' \begin{bmatrix} B \\ 0 \end{bmatrix} F^{-1} \begin{bmatrix} B' & 0 \end{bmatrix} K_s = \dot{K}_s \end{array}\right\} \quad U = U_{p_{OPT}} + U_{s_{OPT}}$$

FIGURE 5. OPTIMAL CLOSED LOOP SYNTHESIS METHOD

An optimal control performance index, J, is defined which includes both manual and automatic control effort as well as closed loop responses. This leads to the solution of a pair of coupled Riccati equations to arrive at the optimal closed loop system. Applying this method as a design approach allows for direct evaluation of trade-offs between manual and automatic control workload in relation to closed loop dynamic characteristics. Furthermore, adjusting the pilot weighting parameters in Schmidt's performance index in accordance with the method of Reference 17 would permit a direct assessment of the resulting closed loop system flying qualities in terms of a predicted pilot-rating.

Onstott's approach to closed loop criteria in Reference 6 is similar to the above in that his results are in terms of closed loop task performance. However, he employs the classical servo pilot model

as the basis for his pilot description. Also, the relationship between task performance and the pilot's evaluation of flying qualities is more direct. While Onstott's pilot modeling approach evolves from the classical single loop tracking model, it results in a nonlinear model with several unique features, as seen in Figure 6.



MULTIAXIS NONLINEAR MODEL

PROPOSED PRECISION TRACKING CRITERION

FIGURE 6. CLOSED LOOP TRACKING CRITERION

Most significant is the explicit modeling of the pilot's decision logic and control switching for multiaxis control. Provision is also made for modeling a side task for considering workload in connection with closed loop flying qualities. Figure 6 also contains a proposed closed loop flying qualities criterion developed analytically using the model. This criterion applies to a precisely defined closed loop task that involves both target acquisition and tracking. It is further defined to be a finite time (5 sec.) task in order to be more realistic. Note that very good correlation is shown between task performance measures and subjective flying qualities evaluation for a wide range of dynamics (Neal and Smith flight test configurations from Reference 14). This approach also appears to have potential for further exploitation.

CONCLUSIONS REGARDING CLOSED LOOP CRITERIA

Closed loop, or pilot-in-the-loop criteria, appear to be appropriate means for specifying military flying qualities requirements consistent with present philosophy and technology. A key element to

success in developing closed loop criteria is our understanding of the dynamic pilot-vehicle system and our capability to properly model those aspects pertinent to flying qualities. We have a relatively good capability with regard to the airplane and automatic stability and control aspects. The pilot element is reasonably well understood for precision tracking tasks. However, the capability for handling more complex or higher order activities is not well in hand. The successful applications of current capability are encouraging, though, and may provide incentive to continue work on the more complex problem, as in Reference 20, for example.

It appears that no one approach can, or should, be chosen as "best" among the ones available. Rather, the advantages and limitations of each should be recognized, and the methods used accordingly. The present emphasis in the military on cost and mission effectiveness, however, would give preference to those methods and criteria that afford the designer the greatest latitude in making system tradeoffs to address these additional constraints. Optimal control theory methods potentially have great advantages in their ability to handle diverse constraints, but the problem of relating the abstract model parameters to physical system characteristics is formidable. This aspect of the optimal control approaches remains the most obvious area for additional research. In fact, the general area of closed loop flying quality metrics offers considerable opportunity for future research.

The planned change in the format of military systems-oriented specifications provides a special impetus to pursue the research topics outlined above. The new format will include a Mil-Prime Standard document and an accompanying Air Force Handbook. The Prime-Standard will contain general statements of requirements, which will then be tailored for each application - including insertion of quantitative requirements - in accordance with the Handbook guidance. The Handbook will contain supporting data and will afford an opportunity to present alternative approaches for design to the requirements and for verifying compliance. Such a presentation could include comparative discussion of alternative approaches.

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SECTION IX

WORKING SESSIONS ON FUTURE DIRECTIONS AND THE MIL-PRIME-STD

WORKING SESSION
THE MIL-PRIME-STD, CONTENTS AND PHILOSOPHY
MODERATOR: David J. Moorhouse

In format this working session was quite unstructured. Typically, a comment from one member of the group would initiate comments from other members. With such a philosophical subject, a consensus of opinion was rarely achieved before moving to a new subject. The following is a summary of these discussions.

A brief initial discussion confirmed the opposition to the proposed method of accounting for the effects of atmospheric disturbances. A possibility would be to put similar requirements in 3.1.10 (Application of Levels) rather than 1.5 (Levels of Flying Qualities); Chalk of Calspan suggests this in a written comment. It was generally agreed that these effects need to be taken into account, but there was no obvious or unanimous choice of how to do it. It was also unanimous that the intent of the revisions should be made clear and to do that a revision of the methods of compliance is needed. The major problem appeared to be the requirement to handle severe disturbance (even though thunderstorm turbulence is currently specified in MIL-F-8785B). A suggestion by Chalk appeared to receive favorable reaction - the numerical requirements should apply only in light and moderate disturbances. The topic ended with a concession that the proposals would be reconsidered.

In discussion of the philosophy of the Prime-Standard, Carlson (Boeing) reiterated his suggestion to organize the Handbook to facilitate the design process. The desirability of feeding back SPO information to the Handbook was raised. This led to the suggestion that this feedback process should be formalized for each procurement.

In discussing items that would be candidates for special consideration and revision in the Prime-Standard it was easy to produce a "shopping list":

higher order systems
lateral-directional requirements in general
methods of compliance
Flight Phase Categories
position control vs precision control
etc., etc.

These items were discussed in varying amounts of detail but the consensus again seemed to be that much work is needed, the answers are not available at present. Again, the philosophy of the specification was discussed, particularly the question of specifying what is required directly instead of indirectly. As an example, if the short-period modal requirements are to ensure acceptable flight path control, then the suggestion is to formulate requirements directly on flight path control. It was noted that prototype programs (the YF-16 and YC-14 were specifically mentioned in the discussion) used a brief performance specification with no flying qualities requirements. Questions were resolved with the SPO engineers as they arose during the

program. On the other hand, an extreme performance requirement such as "putting a certain percentage of bullets in the target on the first pass" would be a problem from design and compliance aspects. It was obvious that defining and achieving acceptable flying qualities of a new airplane system will remain a cooperative effort between the contractor and the government.

WORKING SESSION
ADVANCED AIRCRAFT CONCEPTS AND CONTROLS
MODERATOR: MARTIN T. MOUL, NASA/LaRC

This discussion centered on advanced systems possible with the incorporation of flight computers and fly-by-wire concepts and the applicability (or inapplicability) of MIL F 8785-B to such systems. With the flight computer and avionics capability now available the designer can contemplate a myriad of control system configurations for a given airplane and flight mission. Some significant parameters in advanced systems are command augmentation form (control law defining airplane state being commanded by a controller), cockpit controllers, direct-force-producing controls, feedback loops for control and stabilization, shaping networks, filters, scheduling of gains, filter constants, and control laws as functions of flight condition or task as well as operational considerations such as failure detection, redundancy management, and system reconfiguration. Of the above topics CAS (command augmentation systems), cockpit controllers, feedback loops, and parameter scheduling received the major emphasis of this group. Topics such as pilot-modeling, PIO, and equivalent dynamic systems, though significant, were disregarded because each of these was a major theme of another group.

One dominant recurring conclusion from much of this group's discussion was that advanced systems will at best be described by a high order characteristic equation and will produce output responses indescribable by classical flying qualities. It appears that each candidate system will have to be examined on its own basis with the government and the contractor negotiating specific requirements.

Particular discussion topics and results from this meeting follow:

Controllers - Design and human factors data and requirements are needed for advanced cockpit controllers. For some systems being considered controllers for leading and trailing edge flaps, spoilers, speedbrakes, engine nozzles, etc., as well as for command functions or blending of surfaces are required. As an example advanced velocity control might blend speed brakes with engine throttle to provide a rapid deceleration capability and the designer would require some guidance on desired or required controller characteristics.

CAS Moding - Advanced designs will allow law switching as a function of flight condition. For example a control system might provide pitch rate control at high speed and flight path control at low speed. Requirements are needed for the satisfactory implementation of such capability.

Control margin - The provision of adequate control margin for highly augmented control systems is an obvious requirement. However the vehicle and mission conditions selected as requirements for demonstrating control margin must be carefully selected so as to avoid serious over design in maximum control capability.

Roll rate requirements - The perennial complaint of roll rate requirements being too high was made. Comments were that "maximum roll rates should be restricted to lg flight and certain critical flight conditions rather than require the maximum capability at all points within the operating flight envelope." Or "if the final concern is for trajectory control or some other state, express the requirement in terms of this final state rather than roll rate." For example, direct lift and direct side force control can be used for precision tracking in lieu of rolling and pitching.

In summary it is not possible at this time to establish detailed quantitative requirements for future advanced control systems. The generation of requirements will perforce be a continuing process as experience in the design, development, and analysis of these system is acquired.

Group 3: High-Angle-of-Attack Criteria

Present at this working session were a dozen people, half of whom were from the DoD. The constitution of the group led to little controversy; the numbness of the participants led not so much to consensus as to individual ideas and statements of fact. Therefore the proceedings which follow should only be taken as individual ideas that were raised or commentary on problems.

The first point made was that wind tunnel aerodynamic data needs to be more accurate at high angles of attack (AOA). This would help identify problems in the simulation stage of development. Problems were encountered with this imprecision in the F-16 development.

Probably the most significant part of the session was learning some facts about the F-16 from Mr. Bailey of General Dynamics. A summary of his comments is provided here. The flight control system (FCS) of the F-16 is quad-redundant, fail-safe, fail-operational, analog fly-by-wire. It limits AOA in symmetric flight to 25.5 degrees. At AOA above 29 degrees (in the range where departures and spins can occur), the pilot is taken out of the control loop and anti-spin controls are driven by yaw rate. At negative AOA, the rudder was found to be effective; therefore the yaw damper was considered to be enough augmentation and AOA limiting unnecessary. Angle-of-attack sensors are triple redundant; the FCS selects the middle value. True AOA (in degrees) is displayed in the cockpit. Sideslip is measured but used only in the fire control system. It is realized that any AOA limits can be defeated by a maneuver such as a vertical zoom. Nevertheless,

AOA limiting is relied upon to prevent F-16 loss of control. The extent of its flight demonstration program is still being debated.

The stability and control requirement that was found to be critical in terms of FCS design modifications is unique to that airplane: at load factors of from zero to 0.8 n_L, down to 100 knots airspeed and up to limit AOA, freedom from any departure tendency or undesirable oscillation must be demonstrated in 360-degree rolls. The difficulty encountered with this was the tendency of the basic airplane to pitch up due to inertial coupling at high roll rates. At maximum roll rates the elevator would saturate, allowing the aircraft to depart. Since the test pilots insisted on the validity of this requirement, protests against it were overruled. Although finding a fix was feared extremely difficult if not impossible, one was found. It involves limiting the commandable roll rate to 80% of maximum available at these critical high-AOA conditions. Maximum sideslip generated in this maneuver now amounts to only seven degrees.

Mr. Choo of Northrop was proud of the fact that the F-5 has such good airframe stability that at high AOA they simply turn off the stability augmentation system.

Ken Johnson of ASD (a major contributor to this session) summarized experience with the A-10 stall/spin test program. Departure recovery was emphasized, he said, but still all areas of the stall/spin regime were tested. MIL-S-83691 worked well for them in this respect. Bowman's (NASA Langley) control block served well for spin entry. Johnson commented that there is a lot of data on tape at Edwards if someone were interested in asking the SPO for it.

Don Johnston described STI's upcoming (December 1978) simulation being sponsored by AFFDL and NADC at MACAIR. Some of the maneuvers that will be performed involve closed-loop tasks (such as maintaining pitch attitude) at stall; some cases of instability have been found with stable aircraft when a pilot attempts closed-loop control; coupling of lateral-directional motion with longitudinal is involved. He then described the rating scale that STI has developed for use when the Cooper-Harper scale would yield a ten. STI will be conducting a high-AOA survey of government and industry shortly under AFFDL contract.

The balance of this session report will consist of ideas or comments raised at different points in the discussion.

a. Mr. Jenny of McDonnell suggested adding a requirement to consider departures from abrupt pushovers to large negative AOA.

b. Any airplane can be made to depart by simply running out of airspeed.

c. A required limit on allowable sideslip angle was discussed, but was countered by the need for fuselage aiming. This is a new development; its value at very high AOA has yet to be demonstrated in flight.

d. There is no stated requirement for transient turning ability except in terms of limit load factor and roll performance in response to stick or wheel commands. These can be limited by elevator or rudder power.

e. The technique of unloading g's before rolling was discussed . In any case it seems unsafe to rely on a pilot's doing this for departure prevention.

f. On proposed paragraph 3.4.11 Control Margin, we agreed on the difficulty of stating a requirement of sufficient impact and generality without being unduly restrictive or impossible to meet literally. Tailoring to individual procurements was suggested, but concrete suggestions for solution did not emerge.

g. Why not include simple requirements such as positive $C_{n_{\beta}DYNAMIC}$ (does not include nonlinear or FCS effects) or control authority (directional: enough rudder power to counter a given sideslip; longitudinal: enough elevator power to cancel a given pitch rate) within an aircraft's AOA envelope? These have a place as design criteria but we are required to write specifications in terms of performance rather than descriptive parameters.

h. Why not have a minimum approach speed requirement? Traditionally, this has been left to the aerodynamic performance specification. But stall margin is a valid flying qualities parameter in terms of speed regulation, maneuvering, and gust tolerance. The AMST flying qualities specification contains such considerations.

i. Departure must be prevented; during the approach and landing it does no good to specify recoverability.

j. Based on his F-15 and F-16 experience, Skip Hickey of ASD would like a general requirement for 360-degree fighter roll capability at load factors up to at least 3g.

k. A high-AOA capability is beneficial for quick decelerations

in an air combat maneuver. Good examples are the F-14 and T-38/F-5.

We hope to gather many more ideas such as these to evaluate for the MIL-PRIME-Standard.

ROBERT J. WOODCOCK Moderator	ROBERT B. CROMBIE,1LT, USAF Recorder

Attendance

High-Angle-of-Attack Criteria Session

R.J. Woodcock, AFFDL/FGC, Moderator
C.C. Bailey, General Dynamics/Ft. Worth
D.K. Bowser, AFFDL/FGC
L.W. Brown, NASA/Langley
D.K. Choo, Northrop Aircraft
J.T. Clay, Beech Aircraft
G.K. Hellmann, AFFDL/FGC
R.B. Jenny, McDonnell Douglas/St. Louis
K.L. Johnson, ASD/ENFTC
D.E. Johnston, Systems Technology, Inc.
J.M. Stifel, NADC
R.B. Crombie, AFFDL/FGC, Recorder

REMARKS
R.J. Woodcock, AFFDL/FGC

A. FLIGHT TEST

Available flight test time never has been sufficient for specific demonstration of compliance with every requirement of the flying qualities specification. Further, the trend is to increased cost and increased emphasis on operational-type flying at the expense of demonstrating specification compliance. Thus we must rely increasingly on analysis and simulation, plus flight test techniques such as Twisdale's SIFT. Section 4 of 8785B provides for callout, for each procurement, of the means for demonstrating compliance at each stage of development: analysis, simulation, test.

B. CONTROL MARGIN

I would like to share with you an example from a few years back of the need to assure adequate control authority in an unstable vehicle. After losing a number of aircraft through failure to recover from inverted flight, Mervyn O'Gorman, then head of the Royal Aircraft establishment, led an investigation. Their conclusion is apparent in these two figures. Note the date: January 1919. Relaxed static stability is not a new concept.

C. TURN COORDINATION

The stability-axis side-force equation is

$$n_y = \frac{Y}{mg} = \frac{1}{g} Ur - \cos\theta \sin\phi$$

or

$$r = \frac{g}{V_R \cos\beta} (\cos\theta \sin\phi + n_y)$$

In near-level flight, for tolerable β we have $\cos\theta$ and $\cos\beta$ approximately 1. Thus it is apparent that $n_y = 0$ and $r = g \sin\phi/V_R$ are entirely equivalent definitions of turn coordination. But, as pointed out yesterday, $\beta = 0$ can be a significantly different thing at low speed.

Also, since in a steady turn about a vertical axis

$$r = \dot{\psi} \cos\theta \cos\phi$$

we have

$$\dot{\psi} = \frac{g}{V_R \cos\beta} \left(\tan\phi + \frac{n_y}{\cos\theta \cos\phi}\right)$$

showing two different concepts of turning. Wings-level skidding

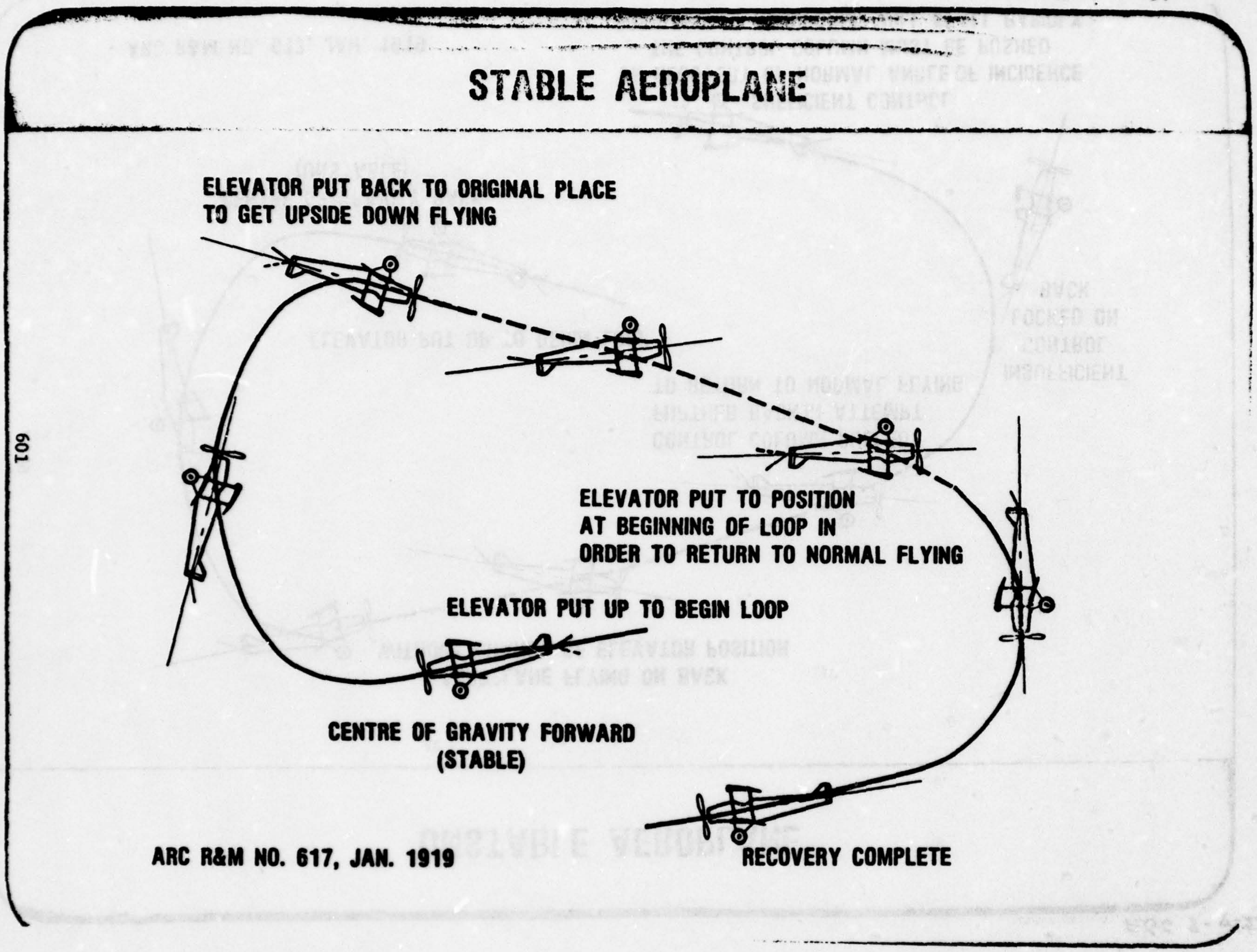
was used before the Wright brothers invented bank-to-turn, and it's once again being considered as an application of direct force control.

D. THE CALSPAN DUTCH ROLL RECOMMENDATIONS

I must confess to being one of those who are confused - as Chick Chalk said yesterday can easily happen - by the graphs and charts in the back of AFFDL TR 72-41. After trying my best to sort out the applicable data points, I fail to see that the Calspan proposal fits the data any better than the present requirements do.

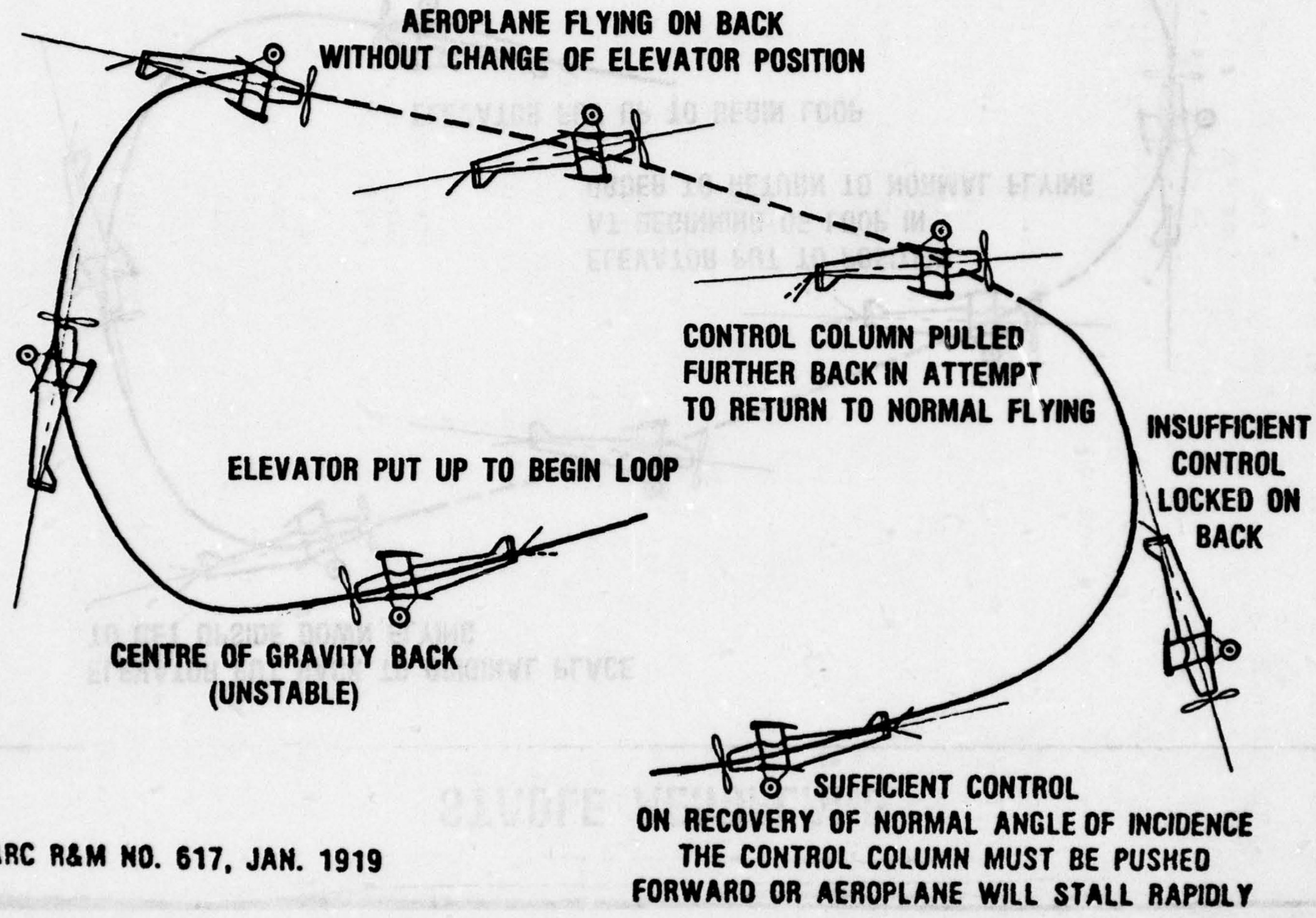
Now, there may be some merit in accuracy and clarity of interpretation of the proposed changes, but we still lack a totally satisfactory requirement on lateral-directional dynamics. I'd like to solicit comments and recommendations on whether or not to adopt these changes.

FGC 4-4-3A


STABLE AEROPLANE
ELEVATOR PUT BACK TO ORIGINAL PLACE
TO GET UPSIDE DOWN FLYING
ELEVATOR PUT TO POSITION
AT BEGINNING OF LOOP IN
ORDER TO RETURN TO NORMAL FLYING
ELEVATOR PUT UP TO BEGIN LOOP
CENTRE OF GRAVITY FORWARD
(STABLE)
ARC R&M NO. 617, JAN. 1919
RECOVERY COMPLETE

FGG 8-8-2

UNSTABLE AEROPLANE



· ARC R&M NO. 617, JAN. 1919

WORKSHOP DISCUSSION: FLIGHT TESTING AND OPERATIONS

MODERATOR: Lt Col Paul J. Shall, 4950 Test Wing

The following is a very brief summary of the topics that were discussed in the Flight Test and Operations Panel on 15 Sept 78 at Wright State. There were two full hours of discussions on numerous issues but I think that only these comments highlight areas of interest to the group. There was good participation from government agencies and contractors. I recommend this panel be continued in future meetings.

1. Flight testing is an extremely important aspect of the total testing and verification process even in view of the other sophisticated methods like analytical methods, ground simulation, and variable stability aircraft for inflight simulation. We must be practical and state which tests to do in flight and which ones to avoid. Several of the proposed Mil-Spec articles are not safe to do in flight. The atmospheric disturbance tests and the PIO verification are cases in point.

2. There is a fundamental difference bewween the conventional open-loop handling qualtties tests and operational tasks. We must be careful in devising operational tests when generating pilot opinion with respect to workload. Make the task representative of operational conditions. We must be careful to project ourselves into the role of a typical line pilot when we give ratings and assess mission capability versus workload. It is extremely difficult to translate pure engineering data into measures of mission effectiveness. If more tests were done in a more nearly typical operational environment, then more valid pilot opinion would result.

3. The government agencies need to be extermely careful in accepting data on the probability of failure in view of the relaxation of requirements authorized in the Mil Spec.

4. Level 2 should be made closer to Level 1 requirements in view of the fact that in the operational environment, only a minor fraction of a pilot's attention can be devoted to compensating for degraded handling qualities. If likely failures result in Level 2 handling qualities then ensure that Level 2 is still fairly good.

5. Workload statistics are very suspect and generate overly optimistic pilot ratings because of the lack of the full operational environment in flight test experiments.

6. The mil spec should offer one way to accomplish a test but should allow variations in techniques.

7. The mil spec is completely ill-equipped to handle direct lift and direct side force as well as sidestick controllers. There is plenty of work being done on some of these topics so the results should be incorporated. Contractors should propose that Independent Research and Development (IR & D) money be spent to develop these technologies into design criteria and mil spec requirements. Work needs to be done on the specifications for cockpit controllers for direct force as this will be the next issue in flight control.

Summary: GROUP 5, PILOT-IN-THE-LOOP CRITERIA DEVELOPMENT

Session Moderator: Frank L. George, AFFDL

In this session, an effort was made to focus discussion on the following three issues:

1) the role of pilot-vehicle analysis in relation to design, criteria and specification requirements;
2) the type, and role, of closed loop criteria appropriate for flying qualities;
3) viable approaches to the Mil Prime Standard philosophy of performance oriented requirements.

Discussion and interchange occurred on a number of points related to the above topics. Only a summary can be presented here even though all the discussion was considered worthwhile and contributed to the consensus reached on the three discussion topics.

The Role of Pilot-Vehicle Analysis

It was readily agreed that pilot-vehicle analysis methods are useful tools in preliminary design and analysis studies. This application includes evaluation of proposed configurations, or competing designs, against established criteria. However, most people were reluctant to commit themselves to rely on pilot-in-the-loop analysis as an analytical means for formally demonstrating compliance with specification requirements. It follows from this also that it is generally not desirable to state criteria in terms of pilot-vehicle analysis parameters (for example, pilot model parameters or performance measures).

The topic of high-order nonlinear aircraft systems was discussed as highly appropriate for application of pilot-in-the-loop methods. One difficulty with such systems is that responses may differ significantly with pilot activity, resulting in flying quality "surprises". Also, several small items may combine to rapidly create an overall bad effect reflected in the pilot's performance or evaluation. Because the characteristics of such systems are input-dependent, it is essential to include the pilot's characteristics for complete analyses.

Closed Loop Criteria for Flying Qualities

The first point discussed under this topic involved developing a common understanding of the meaning of the term "closed loop criteria." It was generally agreed that, for flying qualities, closed loop criteria and pilot-in-the-loop criteria are synonymous. In other words, closed loop flying qualities criteria must address the combination of the pilot and the airplane dynamics as a coupled system. From this perspective, three different ways of quantifying criteria were suggested.

One approach would be to state criteria in terms of desirable, or acceptable, closed loop task performance measures. This approach addresses most directly the end product's capability to complete a mission or task. However, it was pointed out that it is common to experience poor correlation between task performance and the pilot's subjective flying qualities evaluation.

An alternative approach would be to specify closed loop, or pilot-in-the-loop, dynamic characteristics that would assure reasonable task performance. While this approach has much the same weakness as the method above, there are techniques for relating the

pilot's dynamic requirements to his evaluation and standard analysis methods would permit adjusting the airplane characteristics to achieve the best combination.

The second alternative discussed leads directly to a third - defining acceptable pilot dynamic characteristics for achieving desired closed loop dynamics or performance. This approach also suffers from the problem of correlating the pilot's dynamic characteristics with his subjective evaluation. An additional problem, at least for the present, is the limited range of tasks for which pilot dynamics can be reliably defined, or measured.

Thus, the general consensus from the first two topics discussed was that closed loop analysis and closed loop criteria are viable for design purposes but not for military specification requirements. It was recommended that the approach of specifying aircraft dynamics necessary to achieve acceptable task performance be continued. [Ed. note: It is not felt this conclusion was intended to preclude the use of closed loop analysis methods in the development and validation of requirements, however.]

Mil Prime Standard and Performance Oriented Requirements

In view of the consensus on the first two topics, it seemed essential to discuss the philosophy of performance oriented requirements proposed for the new Mil Prime Standard on flying qualities. According to that philosophy, the government standard should describe what the product is supposed to do, and give the supplier responsibility for providing a product with the necessary capabilities. It was pointed out that this philosophy is, in general, consistent with the attitudes of System Program Office management personnel. However, this philosophy becomes more difficult to interpret and apply when considering an item such as flying qualities which is not

like a piece of equipment. One suggested way of considering the difference is to consider the need for defining an acceptable level of quality of performing a task, in addition to the quantitative performance characteristics. In other words, we must continue to consider in some way the Levels of flying qualities as done in the present specification. However, a definitive approach to accomplish this goal was not defined.

Much of the discussion involved peoples' impressions of what the specification (or standard) is supposed to do. In general, it was concluded there are two important aspects of the specification that must be retained. First, it describes the desirable characteristics in general. Second, it provides ways to measure closeness to desired characteristics. These two aspects of the requirements must be in consonance. In other words, satisfying the individual measures of flying qualities should insure meeting the overall goal. In addition to separating good and bad airplanes, the specification should provide for some margin of safety to prevent any surprises in the handling characteristics. This latter objective becomes much more difficult to achieve when considering complex nonlinear aircraft systems.

The general conclusion regarding closed loop task performance requirements in the Mil Prime Standard was that they are undesirable. The principal reason for this attitude was the feeling that such requirements would result in too many side constraints. For example, the test and evaluation method, environment and possibly even the pilot skill level would have to be clearly defined in order to evaluate an airplane against such requirements. Many of these factors are both difficult to define quantitatively and practically impossible to measure and control. Hence, it appears a major area for research is the definition and development of flying quality metrics that reliably correlate closed loop task performance to airplane

system dynamic parameters which can be measured, and controlled in some sense, during design and development.

Attendees at Pilot-in-the-Loop Criteria Working Session

This working session was well-attended by a cross-section of government and industry representatives. Everyone participated and contributed positively to the discussions. Therefore, within the limits of the moderator's memory and note taking capability, the conclusions summarized above are felt to represent the feelings of the entire group. A list of the attendees and their affiliations is given below to give the reader a feel for the backgrounds and viewpoints represented in the group.

Name	Affiliation
Ed Aiken	Army Aeromechanics Lab
Ron Anderson	AF Flight Dynamics Lab (FGC)
Dan Cichy	Rockwell International/Columbus
Bob Fortenbaugh	Vought Corp.
Frank George	Flight Dynamics Lab (FGC)
John Hodgkinson	McDonnell Aircraft Co.
Roger Hoh	Systems Technology, Inc.
William Levison	Bolt, Beranek & Newman
Jerry Lockenour	Northrop Corp.
Walt McNeill	NASA/Ames Research Center
William Pearson	6570 AMRL/HEB
Bruce Powers	NASA/Dryden Flight Research Center
Dave Quam	Univ. of Dayton
Bill Rickard	Douglas Aircraft Co.
Grady Saunders	ARO, Inc.
John M. Schuler	Boeing Co.
Paul Shipley	Rockwell International (Space Shuttle)
Rogers Smith	Calspan Corp.
Hansel Stegall	NASA/Johnson Space Center
Wayne Thor	ASD/YXEF (A-10 SPO)
Frank Wilson	Lockheed-Georgia

SECTION X

SUMMARY AND CONCLUDING REMARKS

SUMMARY AND CONCLUDING REMARKS

The symposium that is the subject of this report was held as a part of an effort to revise MIL-F-8785B, "Military Specification, Flying Qualities of Piloted Airplanes." It was held approximately 12 1/2 years after the last similar conference*, a step in the revision of MIL-F-8785. In this intervening period, MIL-F-8785B was issued in August 1969 and amended in September 1974. Reports have been written validating certain requirements of MIL-F-8785B and not others, suggesting revisions to MIL-F-8785B and even showing how to outsmart MIL-F-8785B. Of particular importance to the authors of the specification is the fact that advancing technology is continually making it more in need of further revision. It is interesting to note then, that some of the discussions at the 1966 conference were repeated in 1978 and some of the same questions remain unresolved.

A. Summary of the Symposium

The symposium papers covered a range of flying qualities topics directly or inderectly related to the specification or the proposed revisions. A total of seven papers directly addressed the subject and these, plus workshop comments and general discussion topics, are discussed in the succeeding two sections. The remaining papers do not necessarily fit in firm categories; however, seven papers suggested new or modified flying qualities criteria. The subjects included high angle of attack, lateral-directional requirements, low speed force gradients, flight path control and the use of equivalent system parameters. Only the last topic was really included in the proposed revisions. Sheer weights of numbers proposing new criteria reinforces the admission that many deficiencies remain and much work is still required.

Five papers are considered to present new results, adding to the total data base. The subjects included higher order system effects (both longitudinal and lateral-directional), sidestick controller characteristics and digital flight control effects. Four papers presented general flying qualities results: an A-10 flying qualities problem that was uncovered by an operational-type mission not by compliance with MIL-F-8785B, flying qualities evaluations of a canard fighter configuration and the Space Shuttle, and a review of ancient but well-used PIO documentation. Finally four papers discussed the Prime-Standard format. A total of eight working groups were convened to discuss the subject topic from a variety of viewpoints. The foregoing is obviously only a very brief summary of the body of this report.

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* "Flying Qualities Conference, Wright-Patterson Air Force Base, Ohio, 5 and 6 April 1966," AFFDL-TR-66-148, December 1966.

B. Discussion of MIL-F-8785B

The papers on MIL-F-8785B presented to the symposium contained comparisons of the requirements with the C-5A/C-141/L-1011; the B-1; and the Advanced Medium STOL Transport (AMST); plus a summary of validation reports for the F-4, F-5, P-3 and C-5A. The trends shown in these papers are that MIL-F-8785B loses validity for large aircraft and for highly augmented aircraft. In the summary paper, only two items were common to all four validation reports. The requirement for phugoid damping was deemed to be too stringent (not addressed in the current revision). There was also a plea for quantitative requirements for flight in turbulence; there was felt to be insufficient data to accomplish this in the current revision effort. The subject is discussed in Volume II of the revision Working Paper. It is of interest to note that for the AMST, MIL-F-8785B was used more than MIL-F-83300. It is planned to incorporate STOL requirements into the future Prime-Standard and Handbook version of MIL-F-8785B.

There was discussion at the symposium about requirements with and without selectable functions engaged (see Carlson's first paper in this report). Section 3.1.5 of MIL-F-8785B dictates that all configurations required for mission accomplishment shall be examined. It does not require Level 1 characteristics with and without all selectable functions. The example used in the reference paper concerned a speed-hold system used to produce Level 1 characteristics. The system is engaged by pilot selection, and it was stated to be obvious to the pilot if the system were not engaged when attempting a STOL landing. The application of 3.1.5 to this example seems perfectly straightforward. If engagement of the speed-hold system is required for Level 1 characteristics then it has to be made a part of the standard landing procedure, as much as putting the wheels down. 3.1.5 does not then require Level 1 characteristics with the speed-hold system disengaged, although possible failures of the system require consideration for other requirements. On the other hand, if use of the speed-hold system is truly a pilot "option," then MIL-F-8785B and prudence would dictate Level 1 characteristics with and without it.

A related discussion concerned switching off augmentation functions for training purposes. The proposed revision to 3.1.1 includes aircrew training as an operational mission, the intent being to recognize that any airplane is used for operational mission training. There may also be training for Failure States and degraded conditions which obviously requires the appropriate worse Level of flying qualities. Paragraph 3.1.5 does not require Level 1 characteristics for these latter selectable configurations.

C. Comments on the proposed revisions*

The symposium contained one formal paper on the proposed revisions (in addition to the one by the perpetrators) plus much discussion, both

* "Proposals for Revising MIL-F-8785B, 'Flying Qualities of Piloted Airplanes,'" AFFDL-FGC Working Paper, February 1978.

in and out of the workshop sessions. There were many comments and suggestions made, and no attempt will be made to editorialize on the majority of these. There was, however, one item that caused strenuous objections and will continue to receive attention, i.e. atmospheric disturbances.

1. Atmospheric disturbances

Environmental conditions, especially atmospheric disturbances, have always been an integral part of flying qualities. The successful design philosophy of the Wright brothers was predicated on minimizing the response to disturbances. To do this they sacrificed stability, with a resulting increase in pilot workload. Report no. 1 of the National Advisory Committee for Aeronautics, "Report on the Behavior of Aeroplanes in Gusts," was concerned with airplane stability and control in gusts. In more recent history, during the effort to produce MIL-F-8785B it was felt necessary to account for the effects of disturbances. Turbulence and gust models were added and some of the requirements were intended to apply in turbulence. The Background Information and User Guide, however, stated: "It was decided, therefore, that turbulence models would be presented in MIL-F-8785B, to be used in any analysis and simulation of flying qualities and ride qualities that the contractor performs." Succeeding discussion concentrated on application to simulation and analyses in general, no specific requirements were imposed. There was no explicit consideration of the effects of disturbances on the Levels of flying qualities.

2. Levels of flying qualities

In the flying qualities community, flying qualities levels commonly are associated with pilot ratings. Pilot (Cooper-Harper) ratings are based on performance and difficulties of the pilot-controlled vehicle in a given task and environment. However, Levels are used to specify values of vehicle parameters acceptable in various flight envelopes, normal and failure states. What determines pertinent tasks and environments is the projected operational usage; the vehicle must be designed to fit these given requisites.

By common observation, pilot rating naturally tends to degrade as the intensity of atmospheric disturbances increases. For conditions encountered not infrequently in a given task or Flight Phase, pilot ratings must be maintained within the appropriate "satisfactory," "acceptable" and "flyable" ranges of Cooper-Harper ratings. Just as clearly, beyond a certain intensity of turbulence it is unreasonable to demand the improvements in vehicle characteristics which would be needed to maintain pilot ratings in the same range (if indeed that were possible). Likewise, there is no desire to lower the numerical requirements as turbulence intensity increases, thus compounding the degradation in pilot rating.

From this discussion it is seen that Levels are associated with the vehicle being procured and its intended missions, whereas pilot

ratings are functions of other factors as well. The flying qualities specification, of course, is not intended to be used for selection of pilots or modification of the environment - these must be accepted as given. Thus the required Levels of aircraft flying qualities are related to stated rational combinations of pilot capability (in terms of workload and pilot-vehicle performance) and atmospheric disturbances.

At the symposium a consensus was evident that these considerations need to be taken into account. It seemed, though, that the number of ways proposed to do this approached or exceeded the number of commenters. We definitely need to do a better job of presenting the concepts individually, one at a time, and then relating them to each other in a manner that is rational, acceptable - and understandable.

One source of difficulty is that we proposed to modify the now-historic singular relationship of 8785B Levels with ranges of Cooper-Harper ratings: POR 3.5 the Level 1 boundary, POR 6.5 separating Levels 2 and 3, and roughly POR 9+ the Level 3 floor. This concept or one like it, is necessary to derive numerical bounds on flying qualities parameters; where insufficient basis exists for such numerical bounds it still provides a frame of reference for qualitative evaluation of aircraft suitability. In concept, few would argue the propriety of that. We do not propose to introduce pilot ratings directly into the specification. For much of the data base, atmospheric disturbances were represented in some manner and degree. Now, however, we see a need to state the obvious - that we neither can nor should force the design of airplanes to have a "satisfactory" rating in severe turbulence. With little data for guidance, the changes proposed are frankly based on intuitive judgment of prospect and need.

Considering the data sources, it seems that the numerical requirements on individual parameters should apply in moderate, if not more intense disturbances. But as was brought out at the symposium, there are practical difficulties with severe disturbances. For one thing, flight situations short of disastrous become harder to evaluate in a number of respects: danger potential, time, limit of tolerance, probability of encounter, estimation of parameters and of the crew's reserve capacity. Also, at some point increasing disturbance intensity will saturate the stability augmentation. That may be no problem with full-authority SAS; but with authority limited in order to bound the effect of hard-over failure in a single-channel system, one would expect noticeable degradation of effective damping ratio, etc. in severe turbulence. We do not want to force unnecessary redundancy or complexity on a designer so it is appropriate to have a qualitative alternative to the numerical requirements for high-intensity disturbance inputs.

3. Proposed requirements

The necessity for providing more explicit requirements to account for the effects of disturbances was acknowledged at the symposium. In

a practical sense this seems quite simple - it is basically a design problem. The designer may minimize response to gusts, add turbulence to a simulation, calculate or simulate response to a wind shear, etc. using rules of thumb. Flight tests are normally scheduled at less turbulent times of the day and certainly not in bad weather. The problem is to formulate this "simple" design requirement in specification format. At the symposium, there was little real disagreement over the announced intent of the revisions proposed by AFFDL/FGC. The comments concerned misinterpretations, both real and anticipated, and counter-proposals for ways to implement the revision. The concensus was to leave the definitions of Levels as in MIL-F-8785B (section 1.5) and add requirements to section 3.1.10, Applications of Levels. In implementing these suggestions, we have found it to be advantageous to define qualitative degrees of suitability to complement the current Levels of flying qualities. There is generally degradation in pilot workload or task performance (i.e. pilot rating) with increasing disturbance intensity, even for an airplane with Level 1 quantitative characteristics. It is now proposed to account for these possible effects by a requirement of the form (as a modification to 3.1.10.1):

Atmospheric Disturbances	Within Operational Flight Envelope	Within Service Flight Envelope
LIGHT TO CALM	Quantitative requirements Level 1; qualitative requirements Satisfactory	Quantitative requirements Level 2; qualitative requirements Acceptable
MODERATE TO LIGHT	Quantitative requirements Level 1; qualitative requirements Acceptable or better	Quantitative requirements Level 2; qualitative requirements Controllable or better
SEVERE TO	Qualitative requirements Controllable or better	Qualitative requirements Recoverable or better

We are still assuming, in application to simulation, that a pilot does not know, or need to know, the actual intensity of the disturbances. His rating is a function of the airplane responses to the disturbances and the workload necessary to achieve the task performance he desires, if possible.

D. Philosophy

In 1966 it was stated*: "The flying qualities specification... is one or more of:

A contractual document
A set of minimum requirements
A design guide
A flight-test standard or guide
A cause of added drag, weight, cost...
A definition of a related subsystem
Assurance, to an extent, of safety, mission capability and good working conditions for the pilot
A research goal

Further evidence of the conflicting requirements is contained in the results of a recent survey** of users of MIL-F-8785B. The document was assessed as a firm specification (48% yes vs 52% no); a design guide (91% yes vs 9% no); and as test and evaluation criteria (87% yes vs 13% no). From the viewpoint of those responsible for MIL-F-8785B it has to be, or form the basis for, all three. It probably follows that it can never be perfectly suited for any one of those uses. MIL-F-8785B will still continue to effect a compromise, and sympathy will be offered to the specialists.

Signal Corps Specification No. 486 is frequently cited as a desirable performance-oriented specification (e.g. F. M. Wilson's paper in this report). The design requirements are explicit in terms of speed, payload, endurance, etc. The flying qualities requirements are implicit in the stated method of demonstrating compliance: "Before acceptance a trial endurance flight will be required of at least one hour during which time the flying machine must remain continuously in the air without landing. It shall return to the starting point and land without any damage that would prevent it immediately starting upon another flight. During this trial flight of one hour it must be steered in all directions without difficulty and at all times under perfect control and equilibrium." It should be noted, however, that this essentially means that the requirement was simply for controlled flight. Although flying was no mean feat in 1907, defining the explicit requirements for many new airplane systems is now a major

* Woodcock, R. J. and Mabli, R. A., "USAF Views on Handling Qualities Criteria," Flying Qualities Conference, WPAFB, Ohio, 5 and 6 April 1966, AFFDL-TR-66-148, December 1966.

** Rediess, H. A. and Shafer, M. F., "Results of Subcommittee D Survey on Future of Research on Flying Qualities and Criteria for Highly Augmented Aircraft," presented to SAE Aerospace Control and Guidance System Committee, October 1977.

task as tactics continually evolve. No longer can we merely demand perfection. G. Brandeau's paper in this report illustrates the flying qualities design problems that resulted from changes in the parameters of a mission task.

An additional problem with truly mission-performance requirements is deciding who flies the airplane. In this context it is interesting to speculate how many modern-day lawyers could gain employment from a dispute of the Signal Corps' requirement that: "It should be sufficiently simple in its construction and operation to permit an intelligent man to become proficient in its use within a reasonable length of time." It is suggested that flying qualities requirements and methods of compliance should properly be negotiated for each new procurement. The future MIL-PRIME-STD and MIL-HDBK is intended to both require and facilitate such tailoring.

1. Flight test

Anxiety was expressed over requirements for which flight testing to demonstrate compliance would be extremely difficult or time-consuming. Requirements related to atmospheric disturbances were of particular concern. However, neither past practice, present procedures nor foreseeable future demands show such difficulty. Flight testing has always been a most pragmatic occupation. That certainly holds with flying qualities. The following discussion attempts to show what reasonably can be expected.

Our first military flying qualities specification, Army Air Forces Specification C-1815, was published in 1943. In all the years since, we believe no flight test program has ever thoroughly checked sensor availability and capability, data recording and reduction equipment, engineering manpower limitations, flight safety considerations, funds availability, aircraft availability, configuration or subsystem changes, urgent problems with other parts of the aircraft, emphasis on operational aspects - the list seems endless. The complexity of a contemporary flight control system itself may preclude flight evaluation of all failure modes.

Currently flight test costs are up, flying hours are down, and emphasis has shifted from engineering evaluation to investigation of conditions approximating operational use. In this climate we must seek optimized flight test techniques to extract the greatest quantity of most-needed flying qualities data in the available flight test time. There is no hope of a flight handling evaluation of the type and scope of AFFTC's Phase IV evaluations of former years. The change is not all bad.

While parameter identification data reduction techniques will never replace flight test time history records of aircraft response

for many analysis purposes, they are seeing more widespread application. As AFFTC has shown here, using appropriate control inputs, data for small perturbations can be accumulated quickly over a large flight envelope for reduction by computer to transfer functions or stability derivatives. Twisdale describes a means of extracting such data from air combat tracking related to the manner in which fighter aircraft are intended to be used. From accurate, well-documented results the aircraft designer's stability and control predictions can be corrected to obtain a validated analytical model.

Those flight tests themselves generate the values of many motion parameters needed to determine MIL-F-8785B compliance. Where they don't an engineer can use the validated model to investigate any aspect of specification compliance at will. With this procedure there are now, of course, many more chances for error along the way. For meaningful results a good deal of coordination is necessary among all those involved in design, testing, evaluation and procurement.

Response to turbulence, gusts, etc. is one example of a type of specification requirement which, though necessary, is practically impossible to flight test. Structural flight loads specifications were the first to put design requirements in such terms. MIL-A-8861 (1960, still used by the Navy) and MIL-A-008861A (USAF) continue use of the time-honored 1-cosine gust which cannot be found at all in flight (especially when looking for one). Compliance with gust-response requirements has always been shown by analysis and ground testing. That holds equally for the statistical turbulence introduced in 1971, by MIL-A-008861A for mission and design envelope analyses. In 1969 MIL-F-8785B introduced flying qualities requirements pertaining to similar gusts and turbulence. In 1975 MIL-F-9490D, the current AF flight control system specification, introduced requirements applicable in atmospheric disturbances that also include wind shears.

As stated in the proposed revision to the British flying qualities specification, Av. P. 970 (RAE Tech. Memo Structures 863, April 1975 Leaflet 600/1), "Compliance with some requirements cannot readily be determined by flight testing... In these cases, compliance can be shown by theoretical calculation or simulation, by agreement with the Aeroplane Project Director, provided that the data used is derived as far as possible from flight testing and provided that some back-up qualitative flying is done; for example, some flying must be done in real turbulence." That approach seems about the best that can be done in flight testing for the effects of atmospheric disturbances. It also greatly expands the ability to show compliance with other flying qualities requirements for which direct demonstration would be very demanding of flight time - such as MIL-F-8785B's roll-sideslip coupling requirements.

E. Concluding remarks

With perspicacity of hindsight it would be possible to improve the symposium - more time allocated to the working sessions and a smaller number of papers. It was judged to be a success from our viewpoint in providing a forum for comment on the proposed revisions and airing many possible misinterpretations of both the existing and proposed requirements. With the forthcoming change in format to the Prime-Standard and Handbook these discussions will assume greater importance. In the near future, it is our intent to hold such symposia more frequently than once every twelve years.

APPENDIXES

APPENDIX A

FLYING QUALITIES SYMPOSIUM ATTENDEES

Aiken, Edwin W.	Army Aeromech Lab
Anderson, Carl	Lockheed Cal.
Anderson, Ron	AFFDL/FGC
Ashbaugh, Wm. H.	ASD/ENFTC
Balsink, Ed.	USAF/ENFTC
Barnhart, Billy	BAR, Inc.
Barry, Jack	AFFDL/FGD
Baxter, Charles	Grumman
Bayley, Jr., C.C.	General Dynamics
Beam, W.R.	OSAF/SAFALR
Bihrle, Bill	BAR, Inc.
Black, Tom	AFFDL/FGC
Brandeau, George	Fairchild Republic
Brown, L.W.	NASA Langley
Brown, Philip W.	NASA Langley
Browne, Jack	AFFDL/FGC
Buckley, J.E.	McAir
Calanducci, Tullio	ASD/ENFTC
Campbell, James E.	Rockwell
Carlson, John	ASD/SD29E
Carlson E. Frank	Boeing
Cathey, John R.	ASD/XRHI
Chalk, C.R.	Calspan
Choo, Duane	Northrop
Cichy, Daniel R.	Rockwell Intl. Col.
Clay, James T.	Beech Aircraft Corp.
Crombie, R.B.	AFFDL/FGC
Cudahy, George, Col.	AFFDL/CC
DiLorenzo, Richard	FTD/TQIS
Emerson, Terry J.	AFFDL/FG
Eresman, Brent	AFFDL/FGC
Flinn, Evard H.	AFFDL/FGL
Fortenbaugh, Bob	Vought, Dallas

Geddes, Norm	SRL, Inc.
Gerken, Gary	ASD/ENFTC
Gertson, W.M.	Gates Learjet
Hamilton, F.	Douglas
Hellmann, Gary	AFFDL/FGC
Hodgkinson, J.	McAir
Hoh, Roger	STI
Iles, Bud	Grumman
Jenny, R.B.	McAir
Johnson, K.	ASD/ENFTC
Johnston, Don E.	STI
Johannes, Robt.	AFFDL/FG
Kandalaft, R.N.	Northrop
Kerzie, D., LTC,	USAF 4950 Test Wg
Kostanty, Ray	Northrop
Kremowski, John	Fairchild Republic
Lamar, W.	AFFDL/CT
Larimer, Stanley	AFFDL/FGC
Lash, Stan	AFFDL/FGC
Lawrence, Tom	NAVAIR
Lemble, E. Robt.	AFFDL/FGL
Levison, Wm. H.	Bolt, Beranek & Newman
Lewis, Tom	AFFDL/FGC
Livingston, E.C.	General Dynamics
Lockenour, J.	Northrop
Markman, S.R.	AFFDL/FGD
Martin, Jim	ASD/XRHI
Martin, Roy	USAF/TPS
Mattes, Bob	AFFDL/FGC
McDonald, E.H.	ASD/ENFTC
McKinney, Royle	NASA/JSC

McNeill, Walter	NASA Ames
Mello, John F.	McAir
Moorhouse, D.J.	AFFDL/FGC
Morse, Chan	USAF 4950 Test Wg
Moul, Martin T.	NASA Langley
Muellner, George	USAF/TPS
Nordwall, Don	Boeing Wichita Co.
Pearson, Wm.	6570 AMRL/HEB
Phillips, D.M.	USAF 4950 Test Wg
Piranian, A.G.	NADC
Pitt, Dale M.	AVRADCOM
Powers, Bruce G.	NASA/DFRC
Quam, Dave	University of Dayton
Raccid, Robt. F.	USAF 4950 Test Wg
Rickard, W.W.	Douglas Aircraft
Rising, Jerry	Lockheed-California
Rowe, R. Kevin	AFFDL/FGC
Rubertus, Duane P.	AFFDL/FGL
Saunders, Jr., Grady H.	ARO, Inc.
Schaeffer, Dwight R.	Boeing
Schilling, Keigh L.	ASD/ENFTC
Schuler, John M.	BMAD
Shall, P.J., L/Col	USAF 4950 Test Wg
Shipley, Paul P.	Rockwell
Smith, Ralph	SRL, Inc.
Smith, Rogers	Calspan
Smith, Stephen	AFFTC
Southern, J.	AFFDL/FGC
Sovine, Don	ASD/ENFTC
Stebe, Jack, Capt	USAF 4950 Test Wg
Stegall, Hansel	NASA/JSC
Stengel, Robt	Princeton Univ.
Stifel, J.M.	NADC

Sweeney, Tim	ASD/ENFTC
Teagor, R.	AFFDL/FGC
Thor, Wayne	A-10 SPO
Twisdale, Tom	AFFTC
Van Vliet, Brian	AFFDL/FGC
Wasicko, Richard	NASA HQ
West, Don E.	Boeing
Wilson, Jr., F.	Lockheed Ga.
Withers, C.C.	Lockheed Ga.
Woodcock, Robt.	AFFDL/FGC
Yeager, R.	AFFDL/MR
Yeakel, Glenn	AFFDL/FGL
Zacarias, Greg L.	Bolt, Beranek, & Newman

APPENDIX B:

AFFTC PARAMETER IDENTIFICATION EXPERIENCE
David P. Maunder, 1st Lieutenant, USAF

The following paper was not presented at the symposium, but was judged pertinent for inclusion in the proceedings in light of discussions concerning flight-test procedures and techniques.

UNCLASSIFIED

AFFTC PARAMETER IDENTIFICATION EXPERIENCE

BY

David P. Maunder, 1st Lieutenant, USAF

Test Engineering Division

Air Force Flight Test Center
Edwards AFB, California

Presented at the Society of Flight Test Engineers National Symposium, October 3, 1978, in Arlington, Texas

AFFTC
PARAMETER IDENTIFICATION EXPERIENCE

1st Lt David P. Maunder
6510 Test Wing, Performance & Flying Qualities Branch
Edwards AFB, California

ABSTRACT

One of the fundamental tasks of engineering and science, is the extraction of information from data. Parameter identification is a discipline that provides tools for the efficient use of data in the estimation of constants appearing in mathematical models of physical phenomena.

The application of parameter identification techniques to aircraft flight testing is simply the process of obtaining quantitative measures of various aircraft characteristics. In general, the parameters may relate to aerodynamic, structural, performance, or other types of characteristics. Typically, the flight-determined characteristics are compared with predicted values to verify or point out deficiencies in the predictions. They are used to substantiate design goals, to assess control system performance, to verify and improve piloted simulators, and to establish design criteria.

This paper presents an overview of AFFTC experience with parameter identification and presents the major results of such application with respect to accuracy, contributions of otherwise unobtainable information, cost effectiveness, and problems encountered.

INTRODUCTION

The Air Force Flight Test Center has been engaged in the determination of the coefficients of "model" equations (stability and control derivatives) which describe the flight characteristics of air vehicles since 1948. In addition, from 1948 to 1961, no less than twenty-two publications on the subject were authorized by the National Aeronautics and Space Agency (NASA, then NACA), the United States Air Force and other agencies (reference 1).

The advantages and benefits of developing a complete mathematical description of an air vehicle's stability and control characteristics has been long recognized by the flight test community. They include (reference 2):

1. Improved verification of performance criteria,

2. Improved dependability of extrapolated flight characteristics.

David P. Maunder, 1st Lt., USAF, Project Manager for Flying Qualities Criteria and Aircraft Parameter Identification Projects, also Chairman of Parameter Identification Working Group.

3. Enhanced system development and optimization of vehicle performance,

4. More accurately represented engineering and operational simulators, and

5. The reduction of the amount of flight test time required to adequately assess the flight characteristics of an air vehicle.

Despite the recognition of the desirability of accomplishing such a task, early parameter identification efforts were limited to the synthesizing of major stability derivatives from data produced by small, linear control inputs. The manual matching of analog computer outputs with flight test traces (analog matching), using linear equations of motion was the most commonly used method of determining these major derivatives (reference 3). Although good results were (and are) achieved, this technique was time consuming and was strongly dependent on the sophistication of the operator. The development of faster, more accurate methods has been accomplished in recent years using improved mathematical techniques well suited for automated computer application. The remainder of this paper concerns the use and application of an automated parameter identification scheme as applied to flight test data at the AFFTC.

The particular parameter identification technique which has been widely used at the AFFTC is the Modified Maximum Likelihood Estimator (MMLE) developed by Kenneth W. Iliff and Lawrence W. Taylor, Jr. of the NASA Dryden Flight Research Center, Edwards AFB, Calif. Experience with this method has been restricted to identification of parameters of linear mathematical models.

Over the past several years this method has been applied on ten major test programs; X-24B, YF-16, YF-17, A-9, A-10, YC-14, YC-15, F-15, F-16, and B-1. The uniqueness of application at the AFFTC has been in its use as the first application of parameter identification as a production analysis tool. The AFFTC has processed more than 1500 maneuvers in limited amounts of time using parameter identification techniques.

PARAMETER IDENTIFICATION EXPERIENCE

The current philosophy of evaluating aircraft at the AFFTC directs aircraft testing efforts toward three major areas; system development, compliance with performance criteria, and minimum testing required. Use of parameter identification techniques has yielded significant improvements toward meeting these goals over previous test methods employed. Due to the nature of this technique, identification of detailed mathematical models is accomplished, thus increasing the information available by which the overall system can be more effectively analyzed and developed.

Test Optimization:

The aircraft flight test maneuvers required for parameter identification are of a different type than is used to obtain the more classical test data and thereby provide an independent test whose results can be

directly correlated with classical testing. The independence of test techniques also allows for an optimum test plan to be developed which utilizes both the new and more classical test methods, thereby minimizing the overall flight test time required. From an implementation point of view, we feel that we have demonstrated significant reductions in the amount of flight test time necessary to define the stability and control characteristics of an air vehicle. During the evaluation of several prototype and production aircraft, records were maintained with respect to the amount of dedicated flight time devoted to classical stability and control maneuvers and those flight hours devoted to parameter identification of STABility Derivative EXtraction (STABDEX) maneuvers.

Table 1

A COMPARISON OF CLASSICAL AND STABDEX
FLIGHT REQUIREMENTS FOR A TYPICAL CONFIGURATION

	Classical	STABDEX
Total Maneuvers	159(89)	117(53)
Total Flight Hours	20.8(11.7)	4.2(1.9)
Maneuvers per Flight Hour	7.6(7.6)	27.9(27.9)
Parameters per Maneuver	2.9(2.9)	10.5(10.5)
Parameters per Flight Hour	22(22)	293(293)

NOTE: Numbers in parentheses are for an idealized flight test program.

Table 1 is a listing of the actual experience during one of these programs in which both classical and STABDEX methods were used. The numbers in parentheses are an estimate of an idealized test program, based on hindsight, which could have been flown to obtain the same data. Two points are obvious. The first is that efficient test planning can effect a significant reduction in flight test time regardless of the analysis method used. The second is that the total flight time can be reduced nearly 75% by the application of STABDEX techniques. It must be noted that the flight time indicated in the tables is not exactly representative of the total flight time required since the evaluation of characteristics such as the variation of pitch control force and deflection with velocity and certain roll performance parameters must be obtained by classical techniques. On the other hand, in a properly implemented active control system, the longitudinal short period frequency and damping ratios can only be quantified by STABDEX techniques.

Accuracy:

The most fundamental and commonly asked question concerning parameter identification techniques is; how accurate are the results? The term accuracy implies an absolute measure of the error between an estimate

and the true value. This of course is impossible to compute since the true value is unknown; however, there are several indicators that have been used which lend confidence and credibility to the results obtained by the application of parameter identification to test data. The three basic indicators of accuracy in the results are; (1) repeatability, (2) correlation, and (3) statistical error analyses.

Figures 1 through 4 shows data which were obtained from an aircraft using the MMLE computer program. Each data point plotted vs angle of attack (α), represents an independent test condition. The data exhibits significant repeatability where data were obtained at similar test conditions. Since these tests were independently conducted and evaluated, the repeatability exhibited lends confidence that the technique yields consistent results. The data presented here is typical of the results obtained on many other test programs conducted at the AFFTC. The vertical lines presented on these plots represent "confidence" levels as computed by the MMLE program. These "confidence" levels will be discussed later.

Figures 5 and 6 are typical examples of the correlation obtained between classical and STABDEX techniques. The data symbols plotted on figure 5 were measurements taken directly from steady-heading sideslip maneuvers, while the faired lines depict data obtained through calculation using STABDEX data. Figure 6 shows similar data; however, this data was obtained over a wide range of flight conditions and is graphic evidence that the STABDEX technique yields nearly identical results as does the classical steady-heading sideslip maneuvers.

A third technique which has been used to validate STABDEX data is the comparison of in flight measured frequency response estimates; that is, measured estimates of the aircraft transfer function, with the Laplace transformation of the equations of motion, where the Laplace transforms are computed using STABDEX data. The result is a direct comparison between measured and computed aircraft transfer functions in the frequency domain. With respect to the dominant model parameters, the results of this technique showed the method to be sensitive enough to easily verify the STABDEX derived model.

In the computational scheme employed in the MMLE program, there exists the capability (under certain conditions and restrictions) to calculate the statistical variance of the estimated parameter value with respect to the true value. This has been shown to be easily accomplished provided one also obtains a measure of the noise associated with the in-flight measured variables. If this can be accomplished, a correction may be made to the "confidence" level which is computed by the MMLE program and which results in an estimate of the variance. The correction factor to be multiplied is N/2B where N is the sample rate and B is the measured frequency bandwidth of the noise (reference 4). This technique has not been employed to date in a production test program; however, we feel that application of this technique will become standard practice. It should be noted that the "confidence" level which is calculated has been used as a relative measure of goodness and parameter sensitivity as can be seen from figures 1 through 4.

Flight Test Parameter Identification Comparisons with Wind Tunnel:

Prior to first flight, wind tunnel estimates remain our primary indicators of how an aircraft will behave. However, there have been several instances where wind tunnel estimates failed to adequately predict aircraft response particularly where initial placard limits were to be established. It is not surprising that differences between wind tunnel estimates and flight test data occur in the rotary derivatives, but, the area of concern, in my opinion, is in the variation experienced in the major derivatives. Figures 7 and 8 are examples of the differences experienced between wind tunnel estimates and flight test data for two major derivatives. Figure 7 shows the wind tunnel estimate for the static margin parameter (C_{m_α}) and shows zero static margin actually occurred near 22.5 degrees angle of attack as opposed to the predicted 14.5 degree angle of attack crossover point. Figure 8 shows the directional stability parameter (C_{n_β}) to be much more stabilizing than predicted. These data have been verified by the techniques mentioned earlier. This discussion is not intended as a condemnation of wind tunnel data or methods, but, rather serves to point out the need for flight testing and increased communication between the flight test and wind tunnel communities.

Parameter Identification and Configuration Effects:

One of the most powerful results of the STABDEX method is in the ability to model relatively small changes in the aircraft's response characteristics due to changes in the aircraft's shape or configuration. Figures 9 and 10 show data obtained for one prototype aircraft. Data shown in figure 9 represents a comparison of static lateral-directional stability characteristics with a change in external stores loading (configuration effect), while the data in figure 10 depicts a change in static lateral-directional stability characteristics with dynamic pressure variations (flexibility effects). It has been our experience that as long as any physical change in the aircraft occurs and produces a measurable change in aircraft response, application of the STABDEX technique has yielded accurate mathematical models of the phenomena.

OTHER CONTRIBUTIONS OF PARAMETER IDENTIFICATION

The examples I have given are only a few of the many which have been encountered in recent years. There are many contributions which have been made through the use of parameter identification which have enhanced our ability to analyze, evaluate, and optimize aircraft performance. One of the advantages in developing a mathematical model of the aircraft is the ability to extrapolate a measured parameter to the next most hazardous flight condition, thereby enhancing safety and mission effectiveness. This has been practiced on more than one test program with great success, particularly near the extremes of the flight envelope. On one prototype test program of a multiengine aircraft, data was obtained during engine-out approaches to stalling maneuvers while maintaining wings level and zero sideslip. The raw measurements of aileron deflection were corrected for aileron required due to rudder deflection yielding the aileron requirement due to an engine-out rolling moment. The measured data were corrected by application of the inflight determined roll control

effectiveness parameters (control derivatives) $C_{l_{\delta_r}}$ and $C_{l_{\delta_a}}$ Figure 11 illustrates the results of those calculations for one configuration tested. The data presented shows how the STABDEX data was utilized to compute and extrapolate the aileron requirements for an engine-out situation to the worst case flight conditions, thereby establishing a margin of safety which would have been difficult or impractical to establish otherwise.

Many evaluations which have been recently accomplished would have been impossible to conduct without the use of parameter identification. This is becoming more and more commonplace with the trend toward active control systems. The two most obvious of all contributions of parameter identification are in the evaluation and optimization of the flight control systems associated with the aircraft, and our ability to provide accurate mathematical models for engineering or operational training simulators.

CONCLUDING REMARKS

The flight test community at the AFFTC has had significant success with the method of parameter identification. Our successes have included the reduction and optimization of flight test programs, improvement in our ability to verify performance criteria, enhanced system development and optimization of vehicle performance, and improvement in the dependability of measured flight characteristics which has also led to more accurately represented simulations. We are committed to the continued use and development of parameter identification techniques, and expect further improvements in flight testing will occur with the development of nonlinear model identification programs and broader applications.

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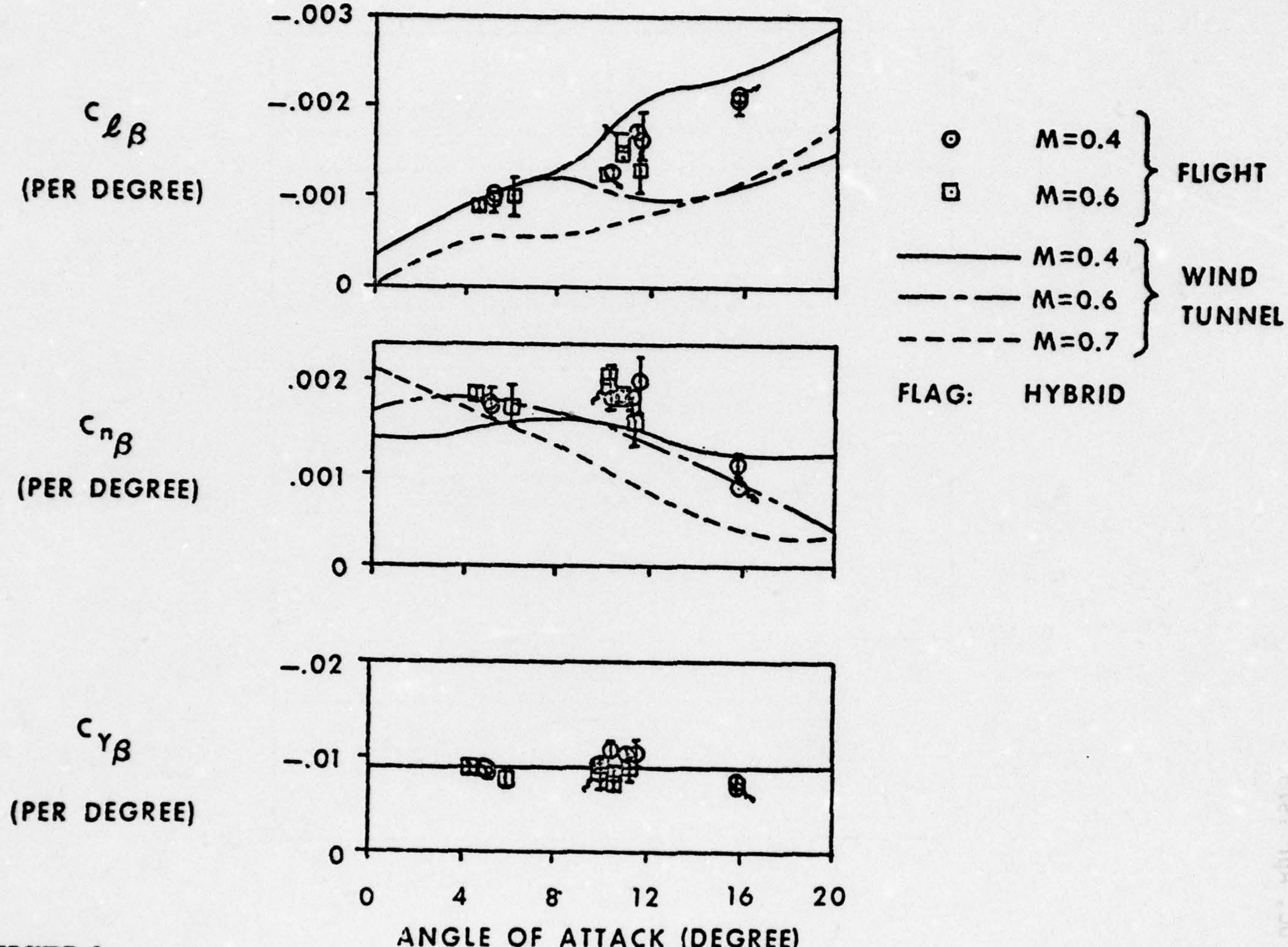
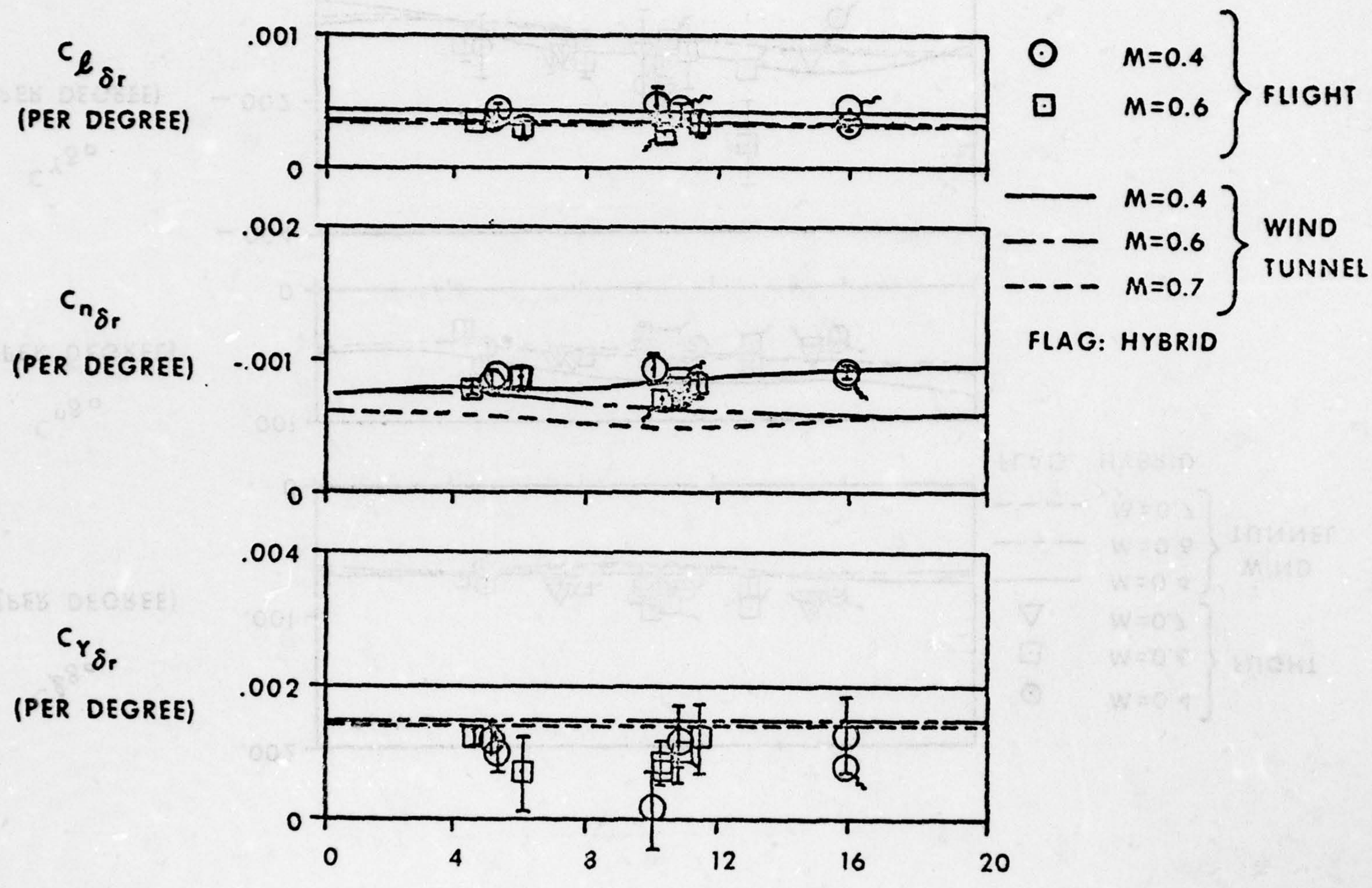
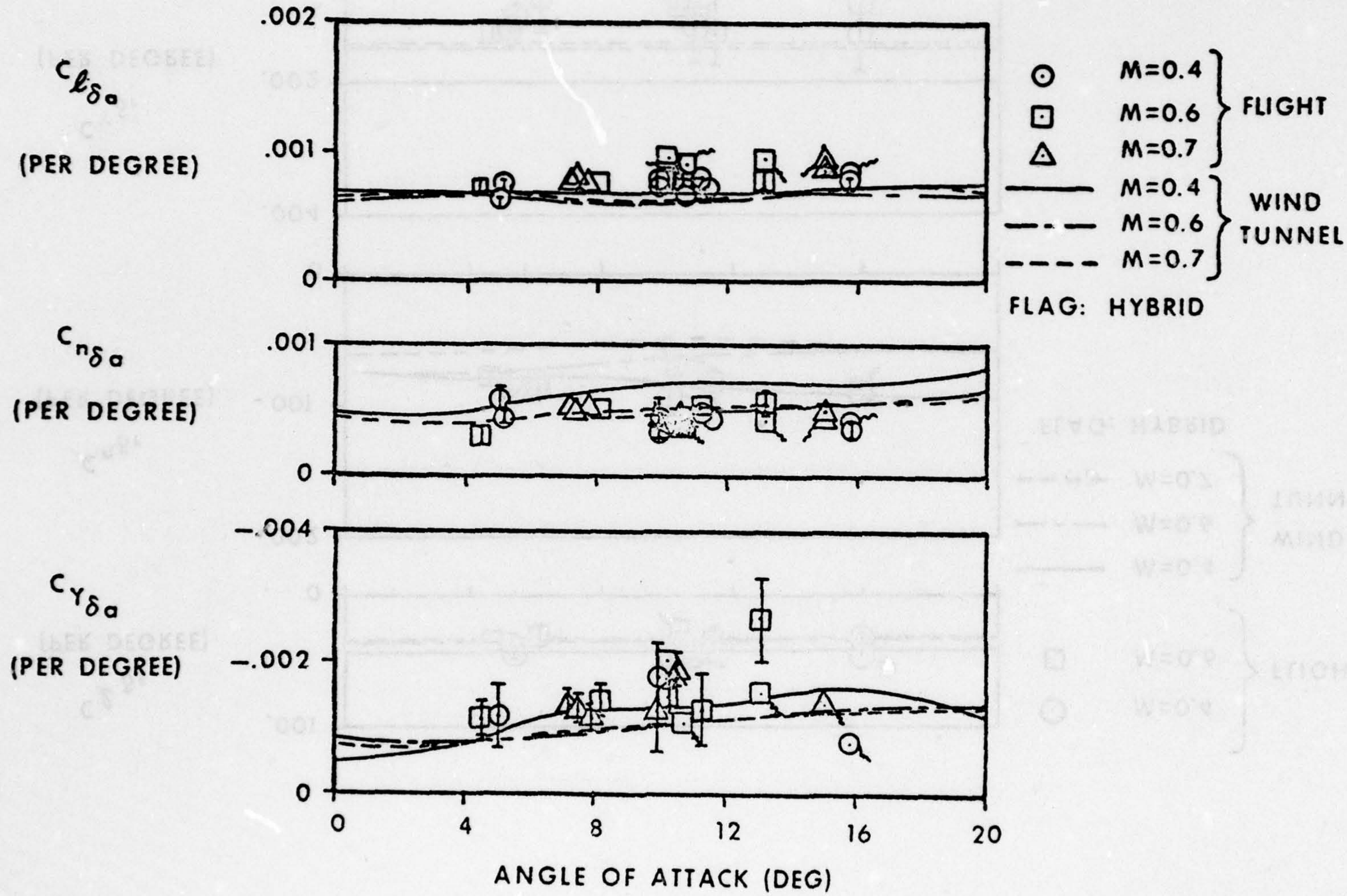

C_{ℓ_β}
(PER DEGREE)
C_{n_β}
(PER DEGREE)
C_{Y_β}
(PER DEGREE)
−.003
−.002
−.001
0
.002
.001
0
−.02
−.01
0
0
4
8
12
16
20
ANGLE OF ATTACK (DEGREE)
M=0.4
M=0.6
FLIGHT
M=0.4
M=0.6
M=0.7
WIND TUNNEL
FLAG: HYBRID

FIGURE 1



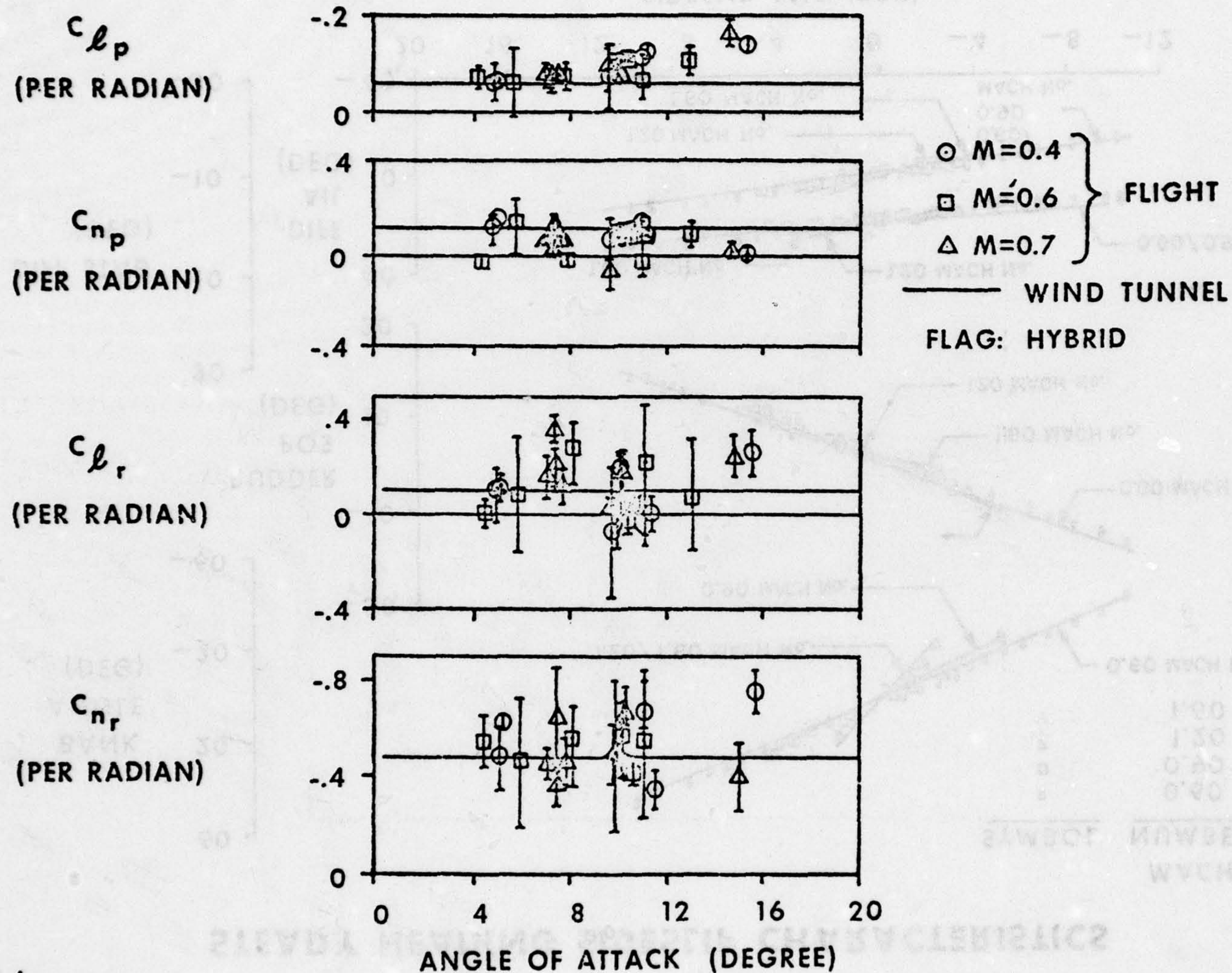
$C_{\ell_{\delta r}}$
(PER DEGREE)
.001
0
$C_{n_{\delta r}}$
(PER DEGREE)
.002
-.001
0
$C_{Y_{\delta r}}$
(PER DEGREE)
.004
.002
0
0
4
8
12
16
20
ANGLE OF ATTACK (DEGREE)
M=0.4
M=0.6
FLIGHT
M=0.4
M=0.6
M=0.7
WIND TUNNEL
FLAG: HYBRID

FIGURE 2



$C_{\ell_{\delta_a}}$
(PER DEGREE)
$C_{n_{\delta_a}}$
(PER DEGREE)
$C_{Y_{\delta_a}}$
(PER DEGREE)
.002
.001
0
.001
0
-.004
-.002
0
0
4
8
12
16
20
ANGLE OF ATTACK (DEG)
M=0.4
M=0.6
M=0.7
FLIGHT
M=0.4
M=0.6
M=0.7
WIND TUNNEL
FLAG: HYBRID

FIGURE 3



C_{ℓ_p}
(PER RADIAN)
C_{n_p}
(PER RADIAN)
C_{ℓ_r}
(PER RADIAN)
C_{n_r}
(PER RADIAN)
-.2
0
.4
0
-.4
.4
0
-.4
-.8
-.4
0
0
4
8
12
16
20
ANGLE OF ATTACK (DEGREE)
⊙ M=0.4
□ M=0.6
△ M=0.7
FLIGHT
WIND TUNNEL
FLAG: HYBRID

FIGURE 4

STEADY HEADING SIDESLIP CHARACTERISTICS

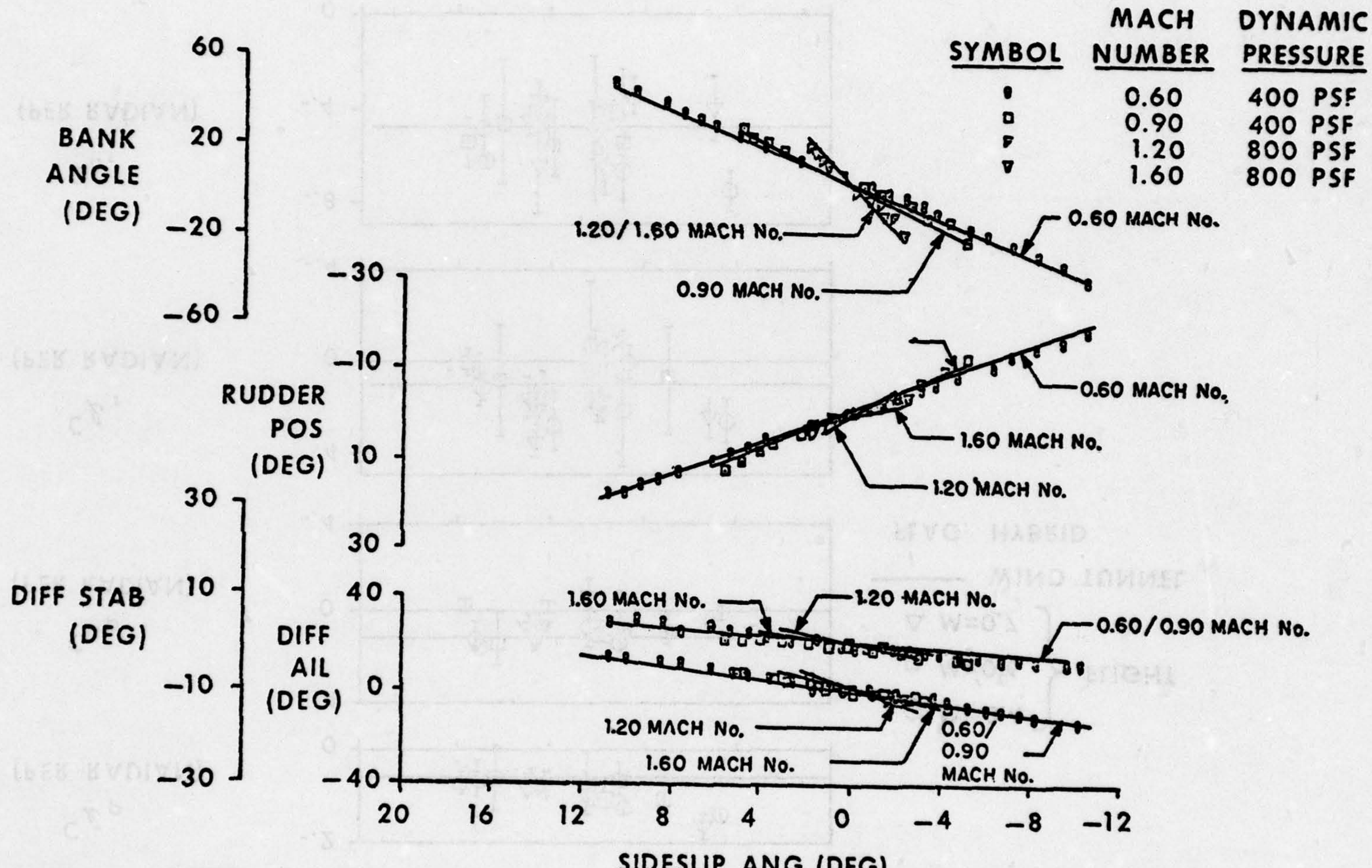
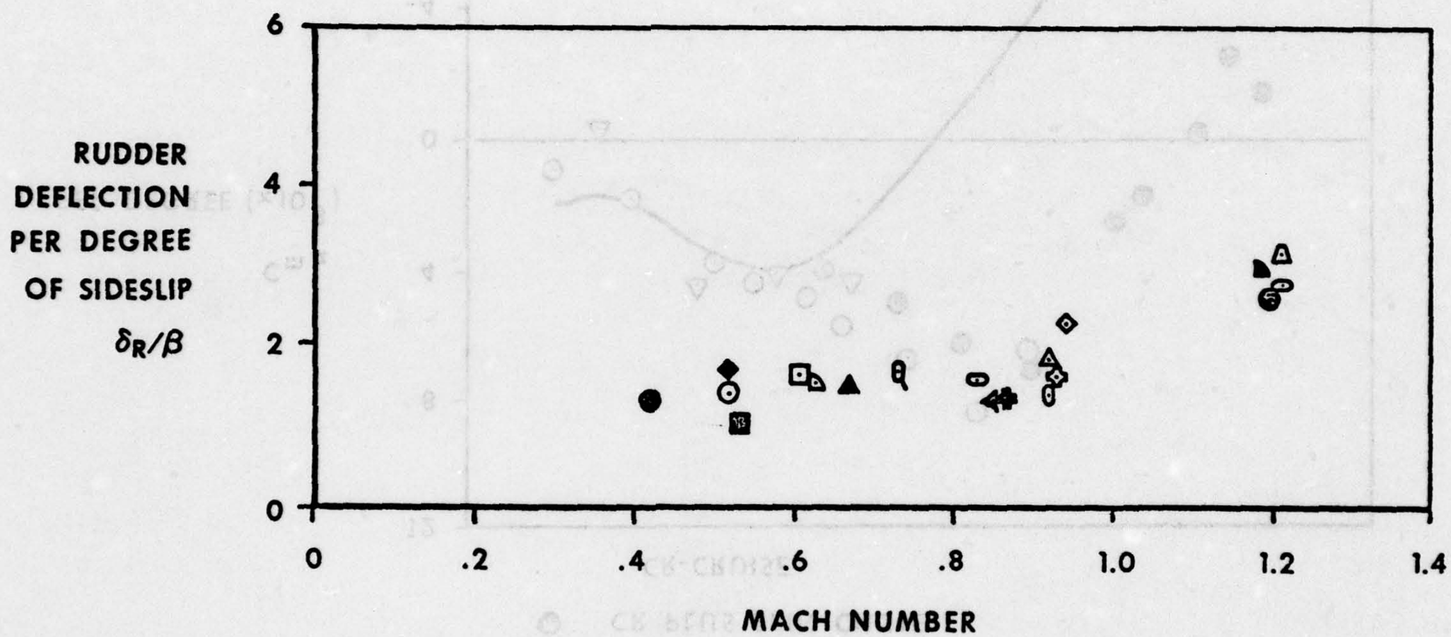


FIGURE 5

NOTES: 1. Data points were obtained from steady-heading sideslip maneuvers.
2. Fairings depict slopes defined by stability derivative data.

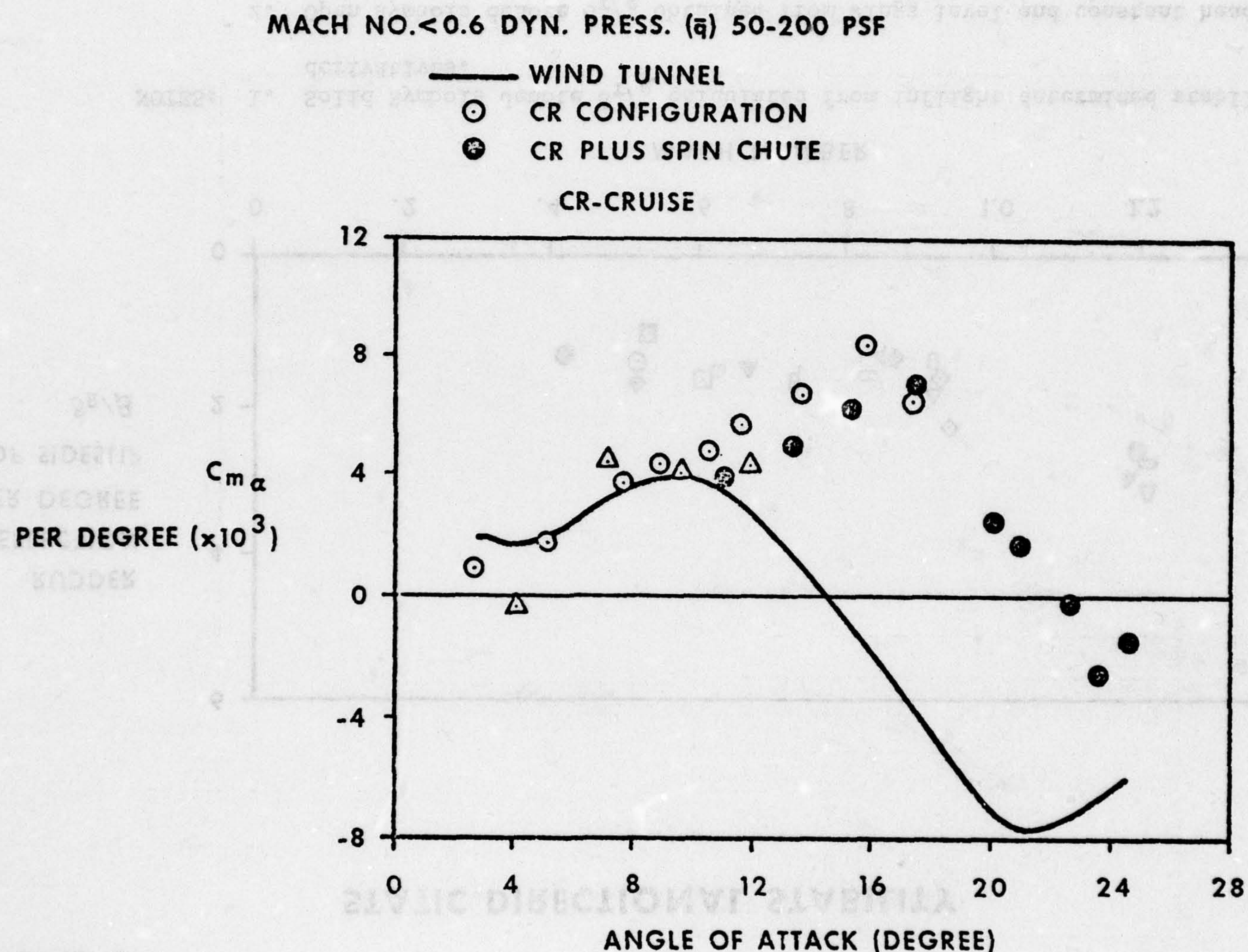
STATIC DIRECTIONAL STABILITY



NOTES: 1. Solid Symbols denote $\delta_r/_\beta$ calculated from inflight determined stability derivatives.

2. Open symbols denote $\delta_r/_\beta$ obtained from wings level and constant heading sideslips.

FIGURE 6



MACH NO.<0.6 DYN. PRESS. (q̄) 50-200 PSF
WIND TUNNEL
CR CONFIGURATION
CR PLUS SPIN CHUTE
CR-CRUISE
C_{m_α}
PER DEGREE (x10^3)
12
8
4
0
-4
-8
0
4
8
12
16
20
24
28
ANGLE OF ATTACK (DEGREE)

FIGURE 7

NOTES:

1. Test data are in the body axis, wind tunnel data are in the stability axis.
2. Fairings from wind tunnel estimates.

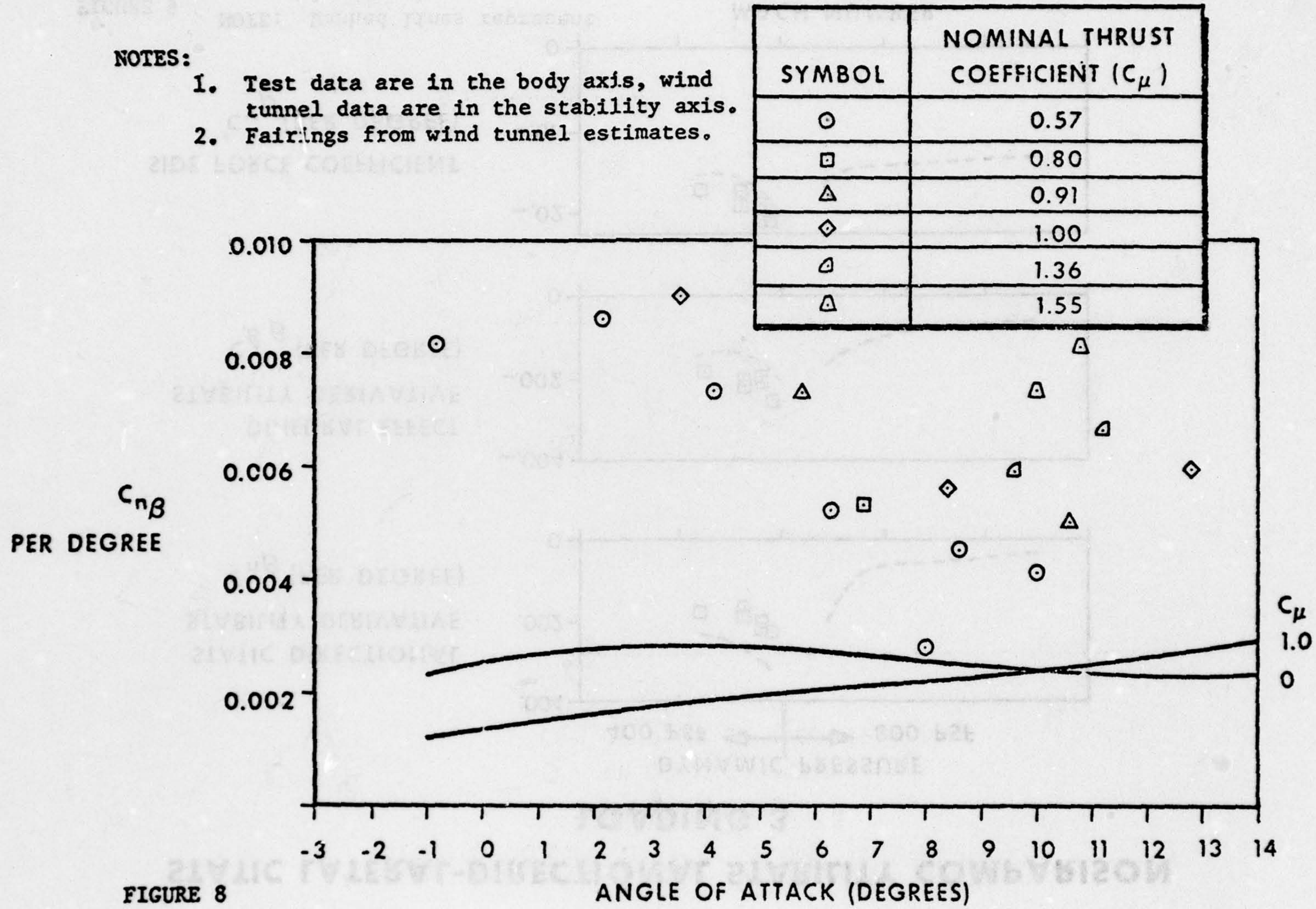


FIGURE 8

STATIC LATERAL-DIRECTIONAL STABILITY COMPARISON
LOADING 3

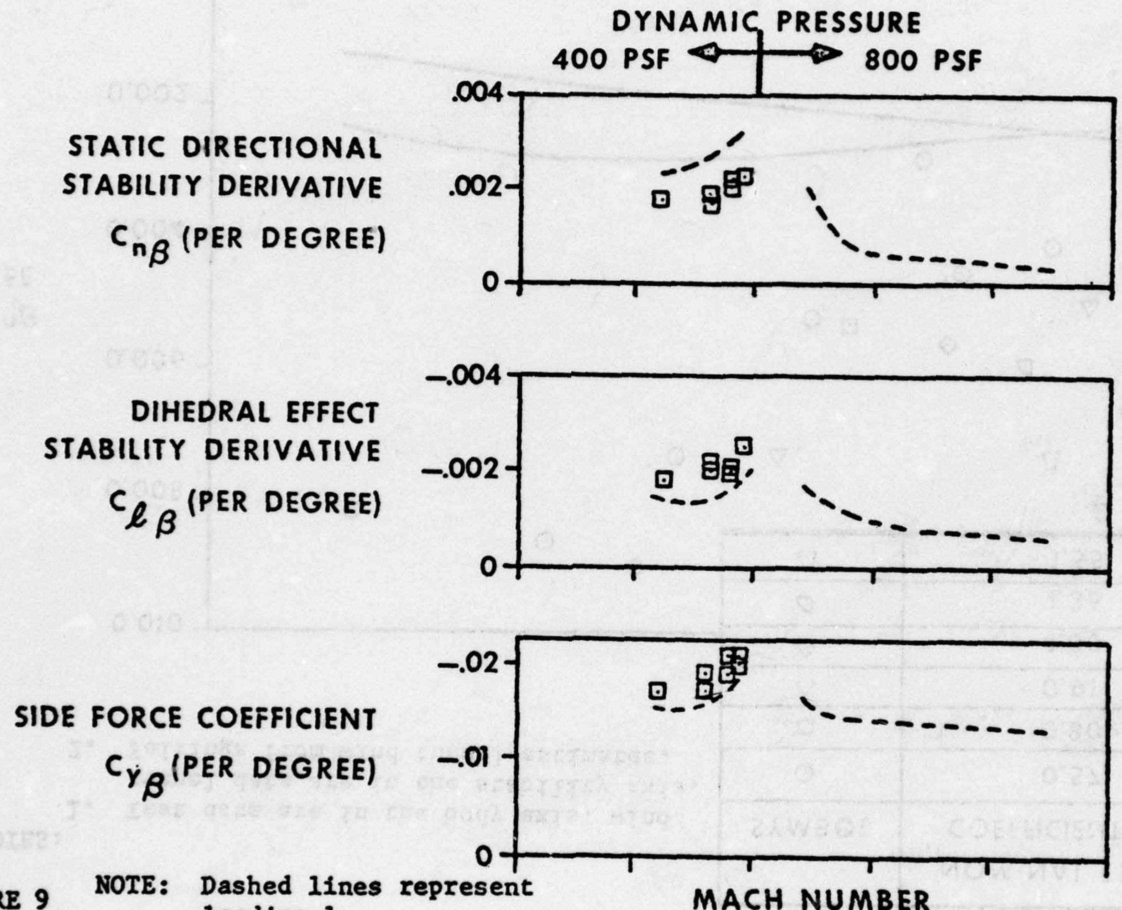
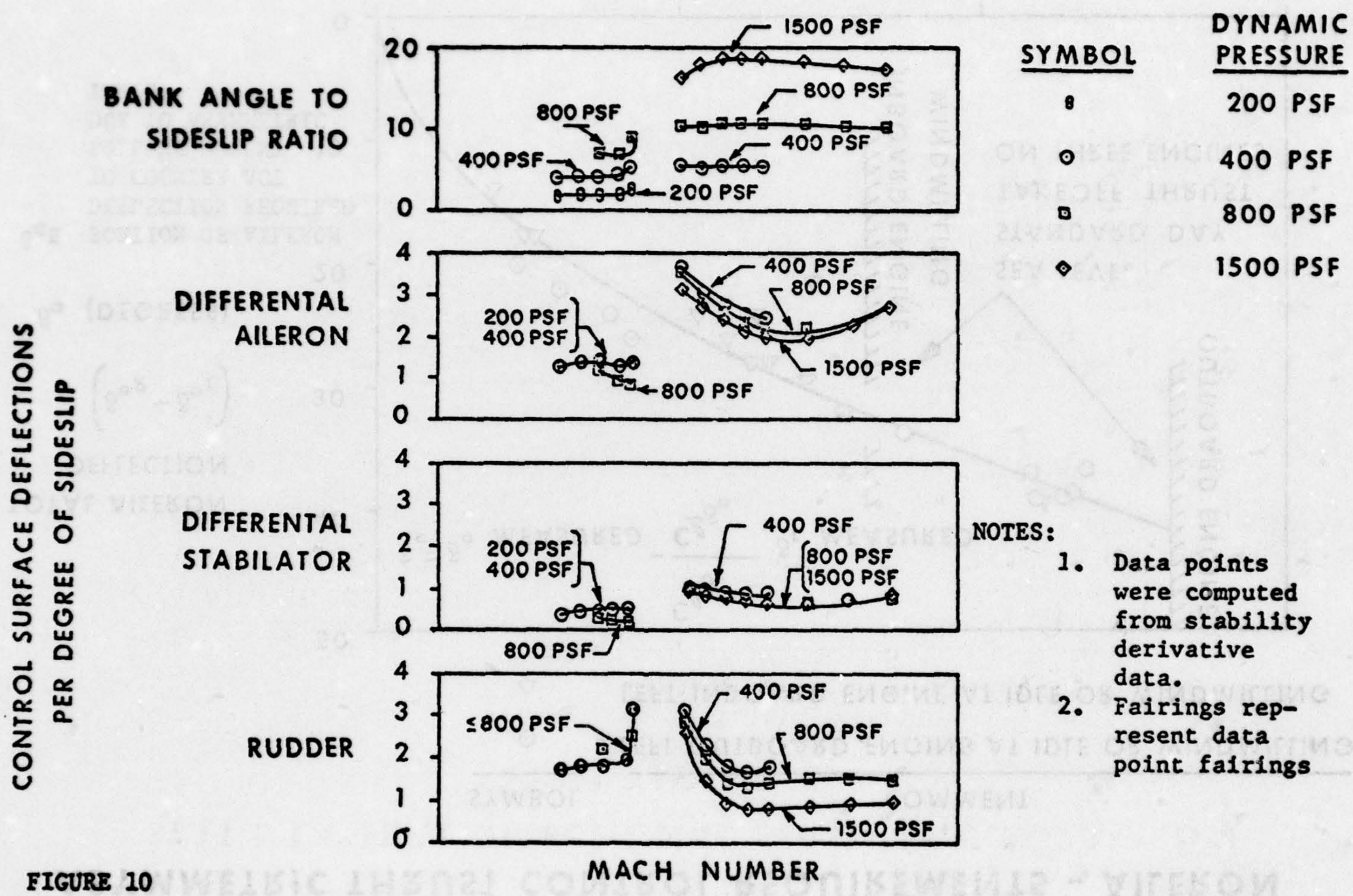


FIGURE 9 NOTE: Dashed lines represent loading 1.

STATIC LATERAL-DIRECTIONAL STABILITY



NOTES:

1. Data points were computed from stability derivative data.
2. Fairings represent data point fairings

FIGURE 10

ASYMMETRIC THRUST CONTROL REQUIREMENTS - AILERON

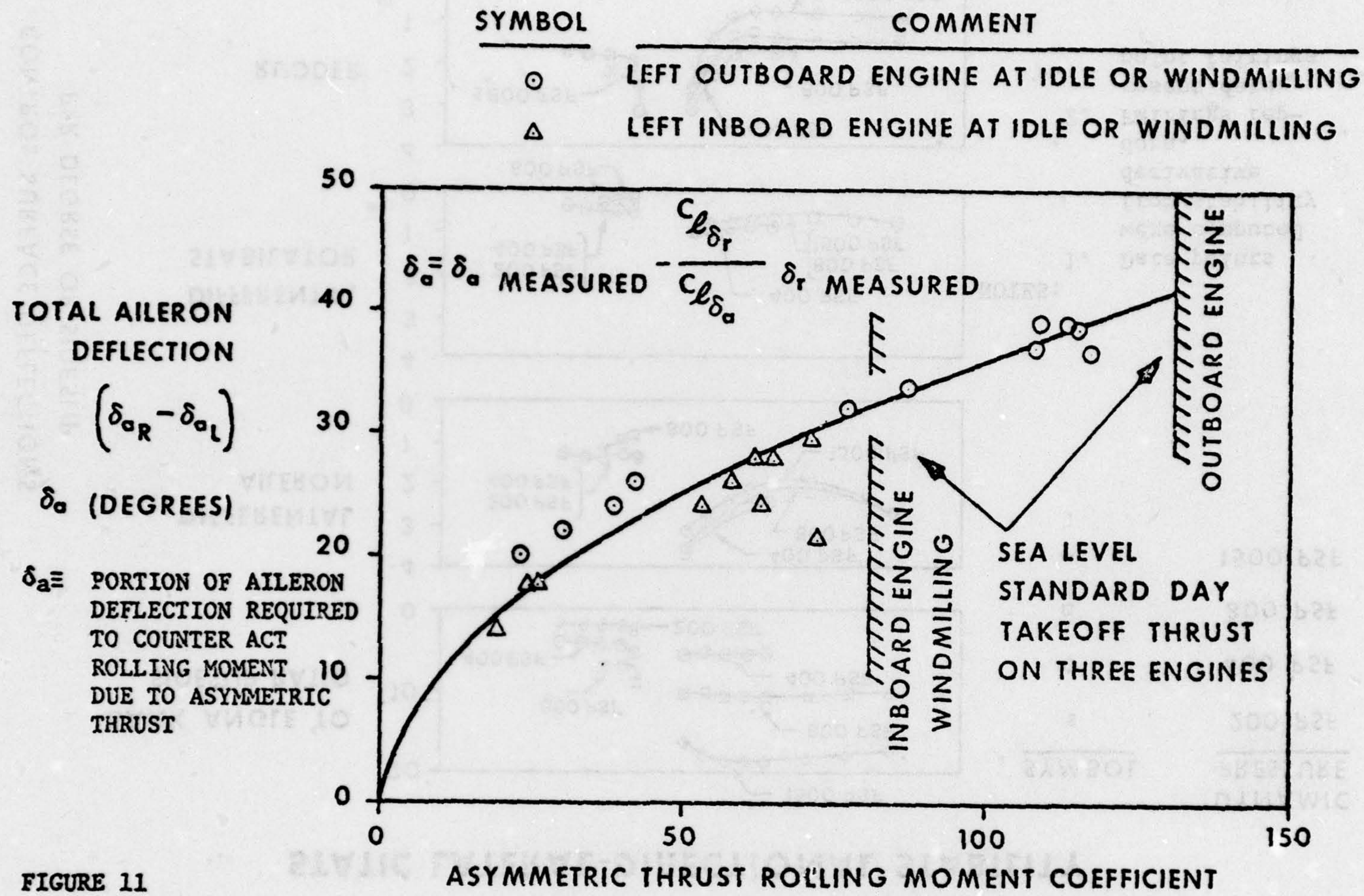


FIGURE 11

*U.S.Government Printing Office: 1979 — 657-002/408

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